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A HINGELESS ROTOR XV-15 DESIGN INTEGRATION FEASIBILITY STUDY

VOLUME I

ENGINEERING DESIGN STUDIES

J. P. Magee

H. R. Alexander

March 1978

Prepared under Contract NAS2-9015

for

National Aeronautics and Space Administration

Ames Research Center

by

BOEING VERTOL COMPANY
A DIVISION OF THE BOEING COMPANY
P. O. BOX 16858
PHILADELPHIA, PENNSYLVANIA 19142

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FOREWORD

The design studies reported in this document were performed by the Boeing Vertol Company for the National Aeronautics and Space Administration, Ames Research Center, under NASA Contract NAS2-9015.

Mr. T. Galloway was the technical monitor for the contract and Mr. J. P. Magee was the Boeing Company program manager.

The following Boeing personnel made significant contributions to the study reported herein.

H. R. Alexander	Technology Manager
F. H. Dean	Design Integration
J. Mack/V. Perillo	Power Train
B. McManus	Flight Controls
C. McCracken/ H. McCafferty	Hub & Upper Controls
C. Class	Blade Design
Y. Badri-Nath/ R. Bainbridge	Airframe Stress
F. Ochs/R. Swinehart	Rotor System Stress
D. Pritchard	Weights
M. A. McVeigh/ C. Widdison	Aerodynamics
C. Chen	Aeroelastics
R. Semple	Propulsion
E. Schaeffer	Acoustics

SUMMARY

In the competition for the XV-15 tilt rotor research demonstrator aircraft contract, the Boeing Vertol design featured a composite material hingeless rotor system. This was sufficiently different from the gimballed rotor of the selected Bell configuration, and had sufficient intrinsic merit to stimulate interest in the idea of flying the Boeing hingeless rotor on the XV-15 at a suitable point in the demonstrator aircraft program.

The subject contract (NAS2-9015), was a design integration feasibility study to investigate what modifications to the basic XV-15 were necessary to accomplish such a flight demonstration and also to explore additional modifications which would exploit the full capability provided by the combination of the new rotor and the existing T53 engine. This implied an upgraded transmission. Other modifications considered desirable were relocation of the engines to a non-tilting position outboard of the rotor, and replacement of the current mechanical controls by a fly-by-wire system.

The cost of the proposed program is minimized by the retention of existing XV-15 systems wherever possible. The cross shafting is unchanged and wing modifications are limited to those required for interfacing with the new nacelle and providing carryover structure to the engine. The existing XV-15 engines are retained eliminating the cost of new engine procurement.

The new transmission and its components are fully interchangeable from right to left hand side, and a high degree of reliability is guaranteed by conservative design and the use of state-of-the-art components which have been the subject of extensive test validation in helicopter applications. A major design objective was ease of inspection and maintenance. This is provided by a modular layout which allows removal of the engine without disturbing other components and by access panels through which engine or power train accessories may be inspected or removed.

The proposed fly-by-wire control system configuration is a triplex self-monitored analog Primary Flight Control System (PFCS) interfaced with a dual analog Stability and Control Augmentation System (SCAS). Both digital and analog computation were considered for both PFCS and SCAS, and the analog was selected on the basis of minimizing program risks related to electromagnetic interference tolerance, configuration control, and development cost.

In addition, the degree of complexity of the SCAS did not present a compelling need for digital rather than analog mechanization. The other major decision was whether the pilot and automatic control channels would share a central computer, or have separate computers to interface with the actuators. In the first case the pilot and stabilization signals are mixed in a central computer which would probably be digital. The second approach provides direct pilot control of the actuators via a relatively simple path. Assuming the aircraft can be flown without augmentation, flight safety is vested in this path; augmentation is via a separate path with redundancy suited to mission requirements. This latter approach is typical of systems found in current aircraft and is recommended for XV-15 applications. The redundancy management scheme uses independent in-line self-monitoring of each channel. This permits the triplex system to be dual fail-functional because each channel independently detects its own failures. This permits simplification of the control logic and prevents any possible failure propagation or electrical interference between channels.

A major milestone in the proposed program is a full scale test in the NASA Ames 40'x80' tunnel. The test article would be the replacement nacelle package and rotor mounted on a semi-span wing. It would include the replacement rotor and nacelle components (e.g., hub, transmission, tilt actuator accessories, etc.), and fly-by-wire actuators for the swashplate. These would be controlled using the aircraft PFCS panel and maintenance unit.

The circuit cards and wiring necessary to control the test rotor would be identical with the final aircraft design.

A successful demonstration in the 40'x80' tunnel would signal the go-ahead for full implementation of the program.

An evaluation of the aircraft resulting from such a program is reported in the text and the data indicate improved air vehicle performance, acceptable aeroeleastic margins, lower noise levels and improved flying qualities compared with the XV-15 aircraft. Inspection of the rotor system data provided shows an essentially unlimited life rotor for the flight spectrum anticipated for the XV-15.

Project planning data to provide the detail design, fabrication and testing of these systems is reported in Volume II.

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1.0 INTRODUCTION

There is an increasing need for V/STOL aircraft for military and special purpose civil applications. Advanced concepts such as the tilt rotor provide improvements in speed, range, and payload, giving increased aircraft productivity, improved fuel economy, and improved mission effectiveness. The current XV-15 research aircraft project is aimed at verifying the feasibility of the tilt rotor concept and the investigation of the basic stability, performance, and handling qualities of the vehicle. The XV-15 currently incorporates a rotor and control system based on 1968 technology. The use of advanced rotor systems with integrated rotor and airplane controls utilizing fly-by-wire concepts will further enhance tilt rotor performance, maneuverability, gust sensitivity, ride comfort and rotor blade life.

The Boeing Vertol Company, both in-house and under contract, has been developing the technology of advanced, composite, hingeless rotor systems as well as integrated rotor/airplane control systems utilizing fly-by-wire concepts. These efforts have included both analytical and experimental studies and have indicated improved capabilities that will broaden the potential application of the tilt rotor concept in the 1980's.

The purpose of this study (performed under NASA Contract NAS2-9015) is to have the Boeing Vertol Company provide the Government with the necessary preliminary design and program planning information for an advanced, composite, hingeless rotor system and fly-by-wire control system on the XV-15 research aircraft.

This study is viewed as the first step of a possible long range three step plan with each possible future step dependent upon the results obtained from the preceding step. The first step is the subject of this report, namely a study to determine the feasibility and practicality of modifying an XV-15 tilt-rotor aircraft with an advanced rotor and other advanced systems, and for preliminary design and associated plans for a 40- by 80-foot wind tunnel investigation. Step two, not part of the scope of work reported herein, would be for detail design, fabrication, and conduct of tests of a full-scale rotor on a wing semi-span in the 40- by 80-foot wind tunnel. Assuming success of step two and funding availability, step three would be for modification of XV-15 aircraft.

The objectives of this study are: (a) to provide preliminary design data showing how the Boeing 26-foot soft-in-plane hingeless rotor system and a fly-by-wire control system could be mounted on the XV-15 tilt rotor research aircraft; (b) to evaluate the performance, flying qualities, noise, and operating limits of the modified aircraft; (c) submit an overall project plan for design, fabrication, wind tunnel testing, and flight testing investigations necessary to support modification of the XV-15 aircraft.

Each of these objectives is the subject of a major section of this report. The preliminary design data are described in Section 2. The technological characteristics are discussed in Section 3.0 and a plan to develop the hardware, and to perform the testing and evaluation is given in Volume II. The notation "HTR-XU-15" refers to the configuration developed in this study.

The rotor system design work included in this report was performed under IR&D. Details of the fly-by-wire control system provided by the Bertea Corporation, General Electric Company, and Honeywell Corporation, are proprietary to the respective companies.

2.0 DESIGN STUDY

This section of the report documents the design integration work performed as Task 1 of the subject contract. In the process of determining the specific configuration, several trade studies and option evaluations were made and this data is reported in Appendix I. The drawing tree for the design study is given in Figure 2.0.

2.1 BASELINE NACELLE ARRANGEMENT

The general arrangement of the nacelle in an inboard profile form is shown in Drawing SK-27244 (Figure 2.1). The general arrangement and dimensional relationship of the nacelle to the XV-15 aircraft are shown in Drawing SK-27243 (Figure 2.2). A basic schematic layout of the system in the nacelle is shown in Drawing SK-27256.

The nacelle assembly consists of two basic bodies alongside each other with the rotor nacelle inboard next to the wingtip and the engine nacelle just outboard. The inboard nacelle is made up of two sections - a tilting rotor nacelle forward and a fixed nacelle afterbody. The outboard engine nacelle is fixed. The tilting rotor nacelle is driven by the same linear hydromechanical actuator mounted in the wing in the same manner as the current XV-15, and moves the nacelle through the same 95° total angular range from hover to cruise flight. Because of the difference in the new nacelle contour, a new tilt actuator fairing will be required, however. The HTR nacelle is mounted to the wing in a manner generally similar to the

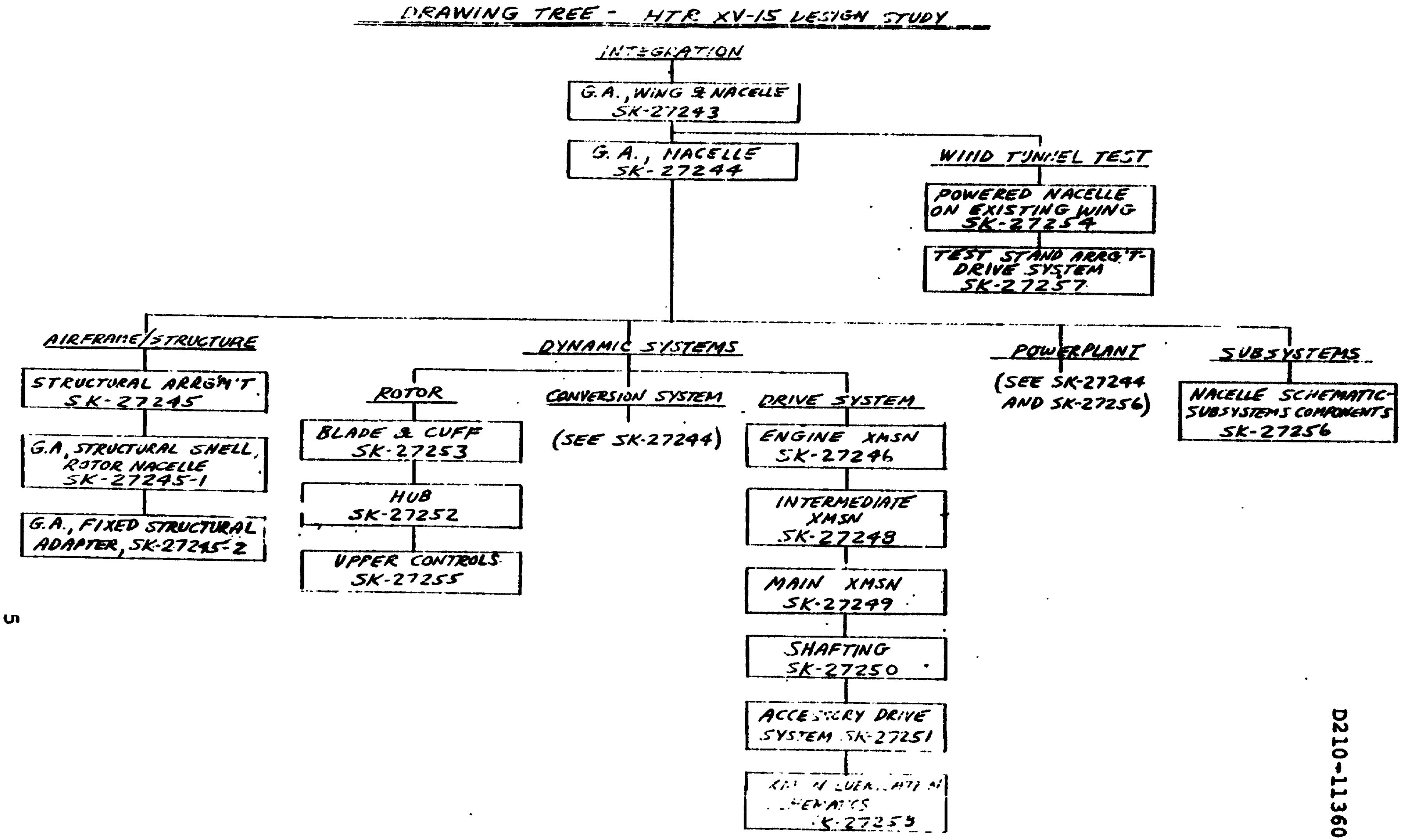
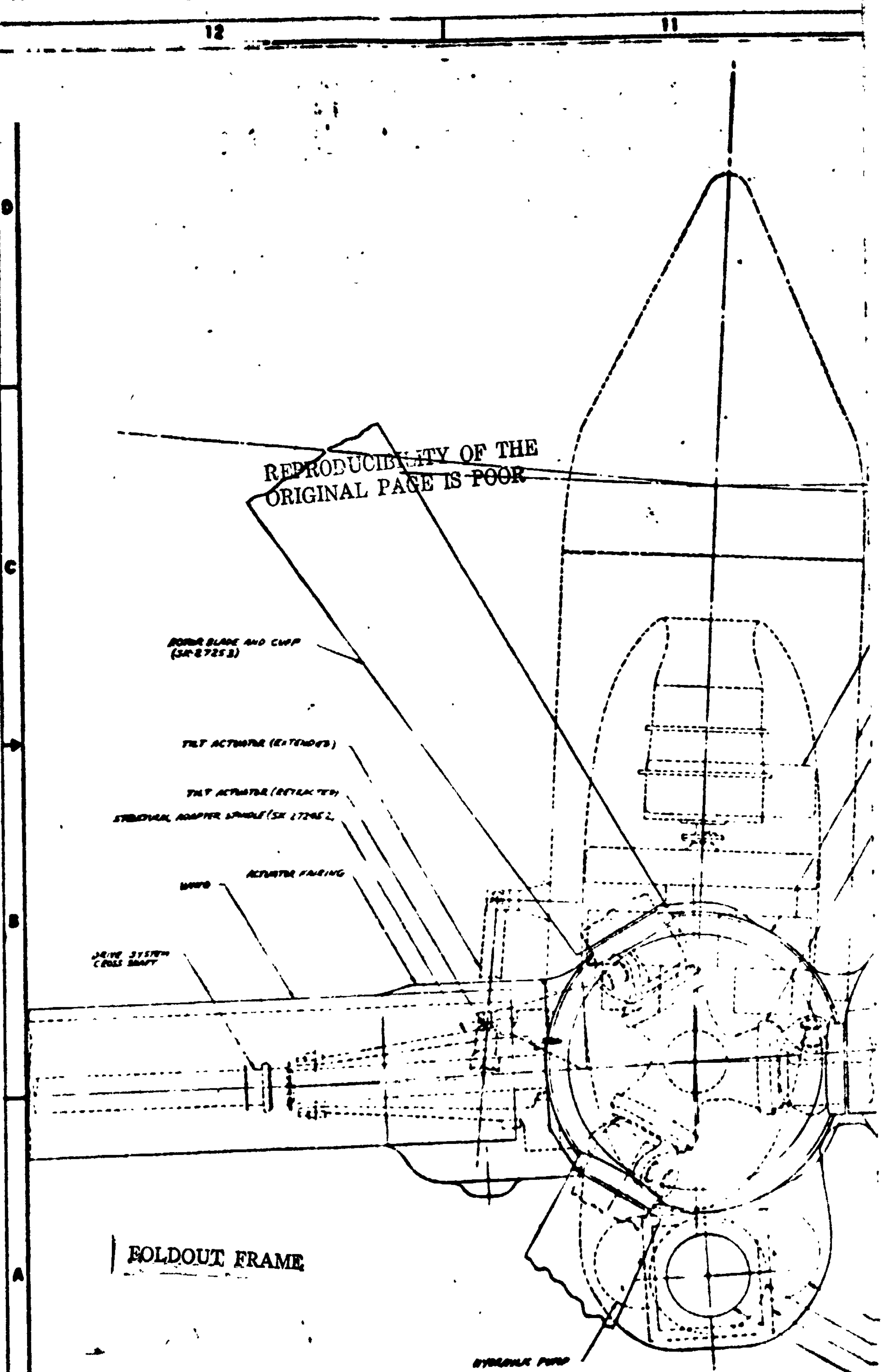
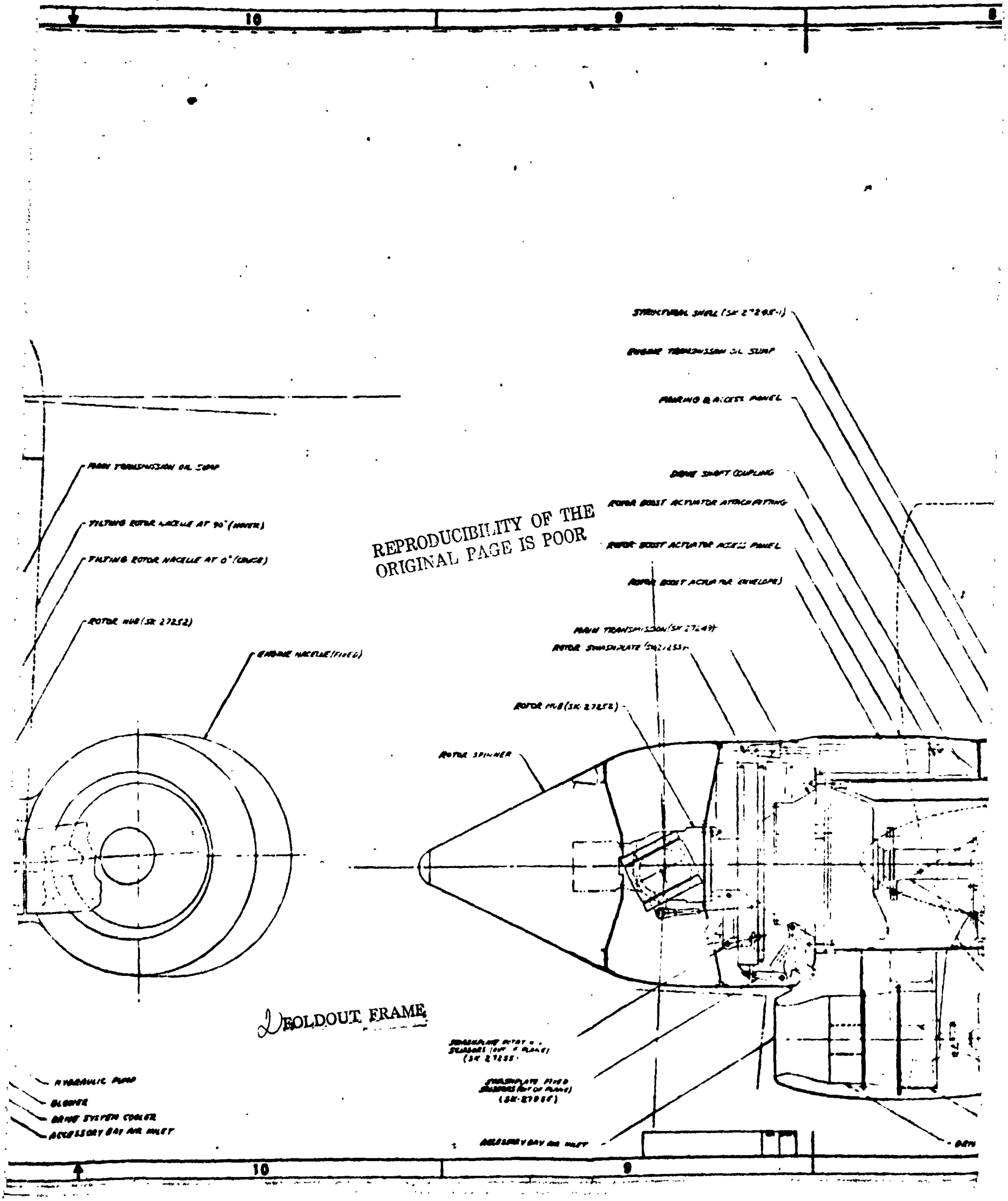


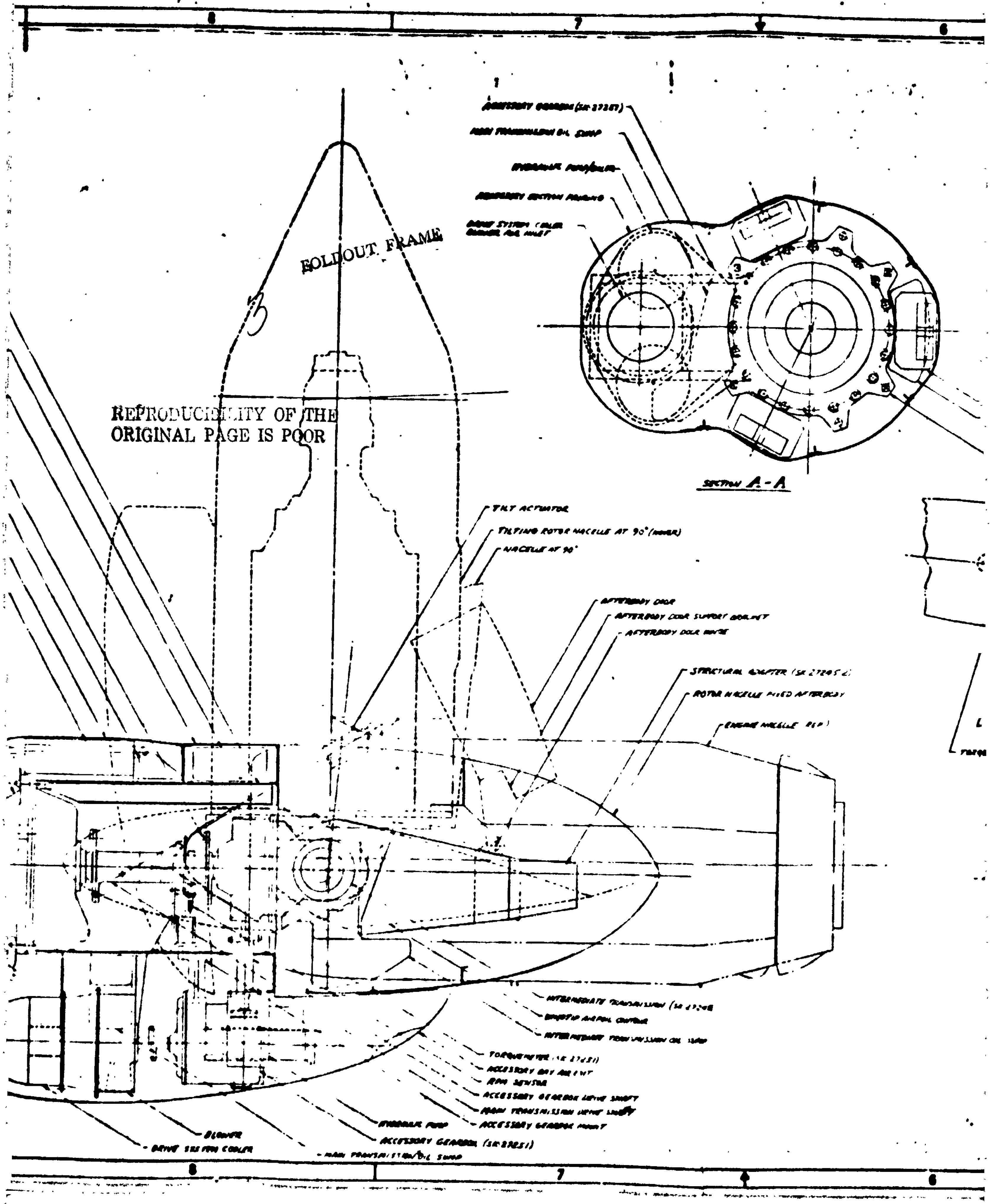
FIGURE 2.0. DRAWING TREE

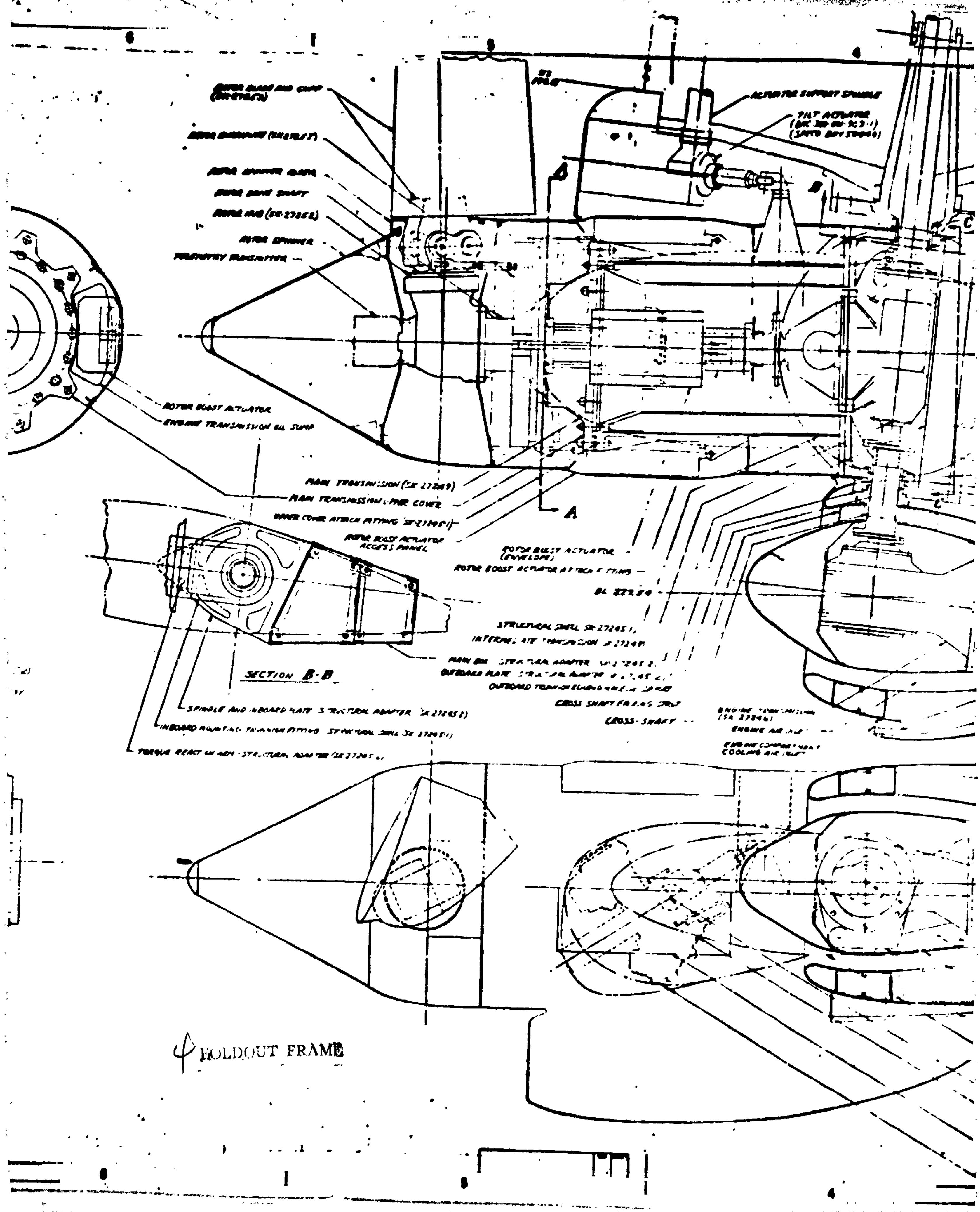


FOLDOUT FRAME

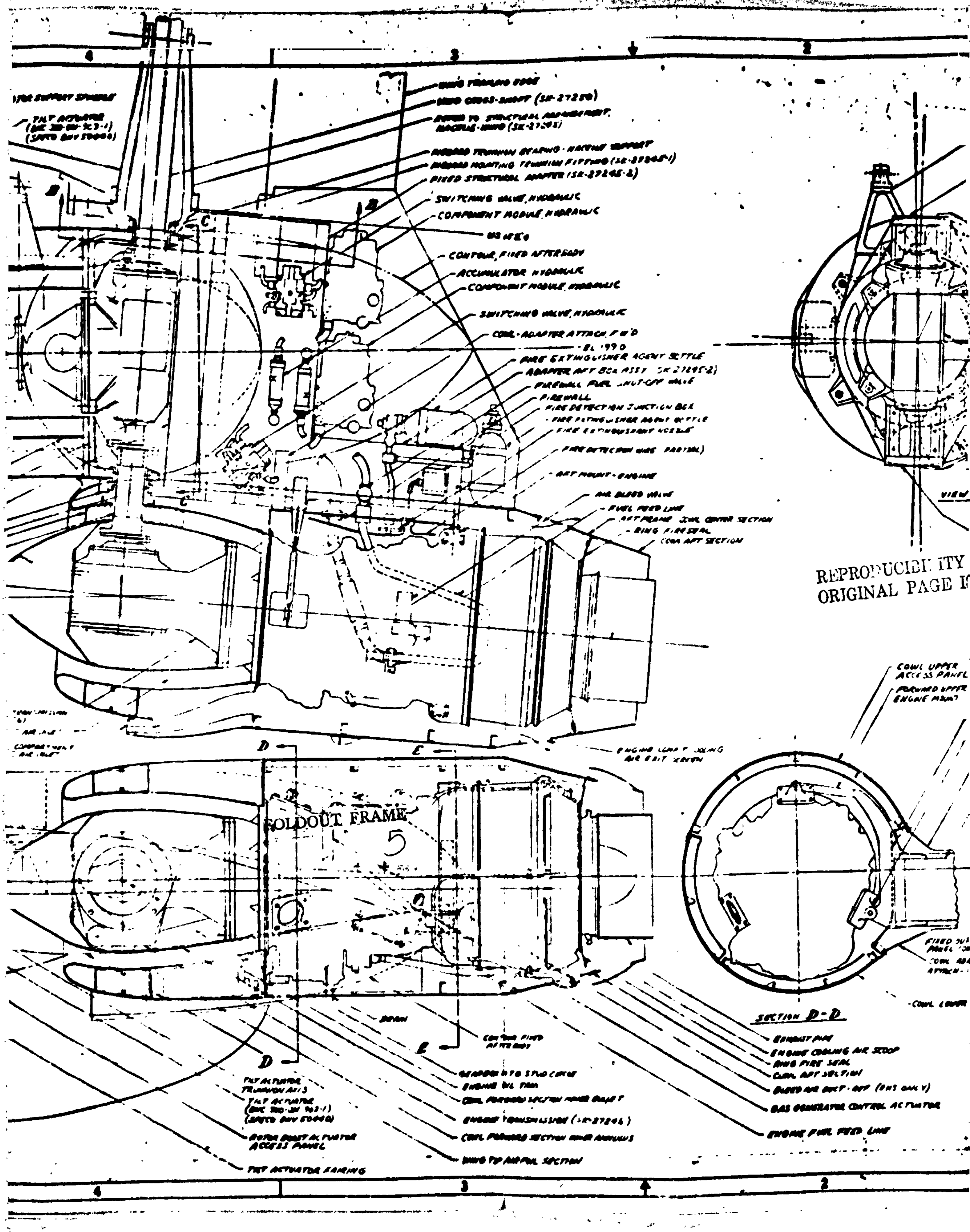
FIGURE 2.1. GENERAL ARRANGEMENT - NACELLE - HTR XV-15


STRUCTURAL SHELL (SK 27295-1)
ENGINE TRANSMISSION OIL SUMP
FAIRING & ACCESS PANEL
DRIVE SHAFT COUPLING
ROTOR BOOST ACTUATOR ATTACH FITTING
ROTOR BOOST ACTUATOR ACCESS PANEL
ROTOR BOOST ACTUATOR ENVELOPE
MAIN TRANSMISSION (SK 27249)
ROTOR SWASHPLATE (SK 27255)
ROTOR HUB (SK 27252)
ROTOR SPINNER
MAIN TRANSMISSION OIL SUMP
TILTING ROTOR NACELLE AT 90° (HOVER)
TILTING ROTOR NACELLE AT 0° (CRUISE)
ROTOR HUB (SK 27252)
ENGINE NACELLE (FIXED)
HYDRAULIC PUMP
BLOWER
DRIVE SYSTEM COOLER
ACCESSORY BAY AIR INLET
ACCESSORY BAY AIR INLET

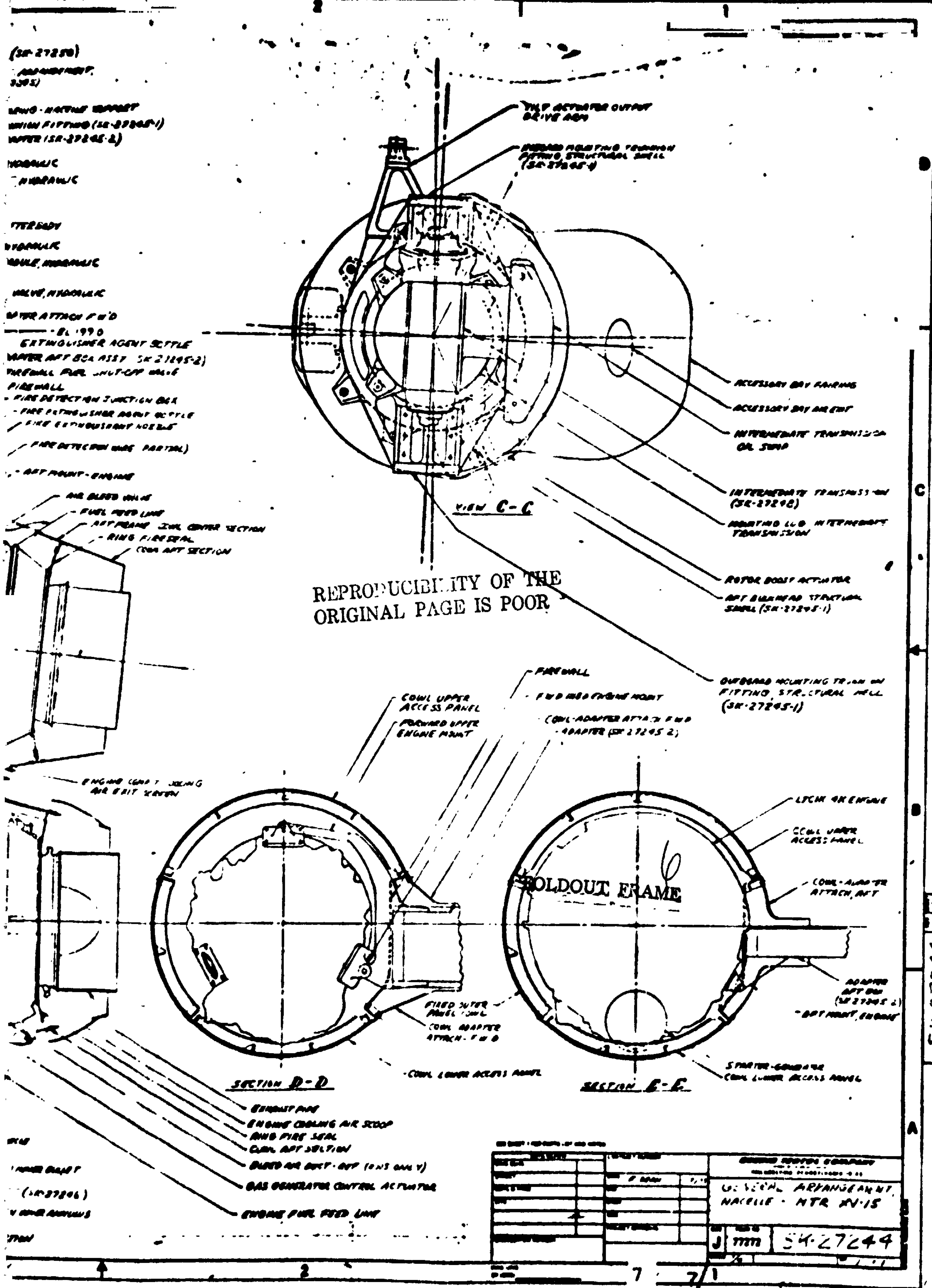

SECTION A-A
TILT ACTUATOR
TILTING ROTOR NACELLE AT 90° (HOVER)
NACELLE AT 90°
AFTERBODY DOOR
AFTERBODY DOOR SUPPORT BRACKET
AFTERBODY DOOR HINGE
ROTOR NACELLE FIXED AFTERBODY
WINGTIP AIRFOIL CONTOUR
INTERMEDIATE TRANSMISSION OIL SUMP
ACCESSORY BAY AIR EXIT
RPM SENSOR
ACCESSORY GEARBOX DRIVE SHAFT
MAIN TRANSMISSION DRIVE SHAFT
ACCESSORY GEARBOX MOUNT
HYDRAULIC PUMP
BLOWER
DRIVE SYSTEM COOLER
MAIN TRANSMISSION OIL SUMP



FOLDOUT FRAME



WING TRAILING EDGE
WING CROSS-SHAFT (SK-27250)
SWITCHING VALVE, HYDRAULIC
COMPONENT MODULE, HYDRAULIC
CONTOUR, FIXED AFTERBODY
ACCUMULATOR, HYDRAULIC
COMPONENT MODULE, HYDRAULIC
SWITCHING VALVE, HYDRAULIC
FIRE EXTINGUISHER AGENT BOTTLE
FIREWALL FUEL SHUT-OFF VALVE
FIREWALL
FIRE DETECTION JUNCTION BOX
AFT MOUNT - ENGINE
AIR BLEED VALVE
FUEL FEED LINE
RING FIRESEAL
COWL AFT SECTION
COWL UPPER ACCESS PANEL
FORWARD UPPER ENGINE MOUNT
FOLDOUT FRAME
5
SECTION D-D
EXHAUST PIPE
ENGINE COOLING AIR SCOOP
RING FIRE SEAL
COWL AFT SECTION
GAS GENERATOR CONTROL ACTUATOR
ENGINE FUEL FEED LINE
DRAIN
CONTOUR FIXED AFTERBODY
ENGINE OIL TANK
ENGINE TRANSMISSION (SK-27246)
WING TIP AIRFOIL SECTION
TILT ACTUATOR TRUNNION AXIS
ROTOR SHAFT ACTUATOR ACCESS PANEL
TILT ACTUATOR FAIRING


VIEW C-C
ACCESSORY BAY FAIRING
ACCESSORY BAY AIR EXIT
INTERMEDIATE TRANSMISSION OIL SUMP
INTERMEDIATE TRANSMISSION (SK-27292)
ROTOR BOOST ACTUATOR
OUTBOARD MOUNTING TRUNNION FITTING STRUCTURAL SHELL (SK-27245-1)
FIREWALL
COWL UPPER ACCESS PANEL
FORWARD UPPER ENGINE MOUNT
SECTION D-D
SECTION E-E
EXHAUST PIPE
ENGINE COOLING AIR SCOOP
RING FIRE SEAL
COWL AFT SECTION
GAS GENERATOR CONTROL ACTUATOR
ENGINE FUEL FEED LINE
GENERAL ARRANGEMENT NACELLE - MTR XV-15
SK-27244

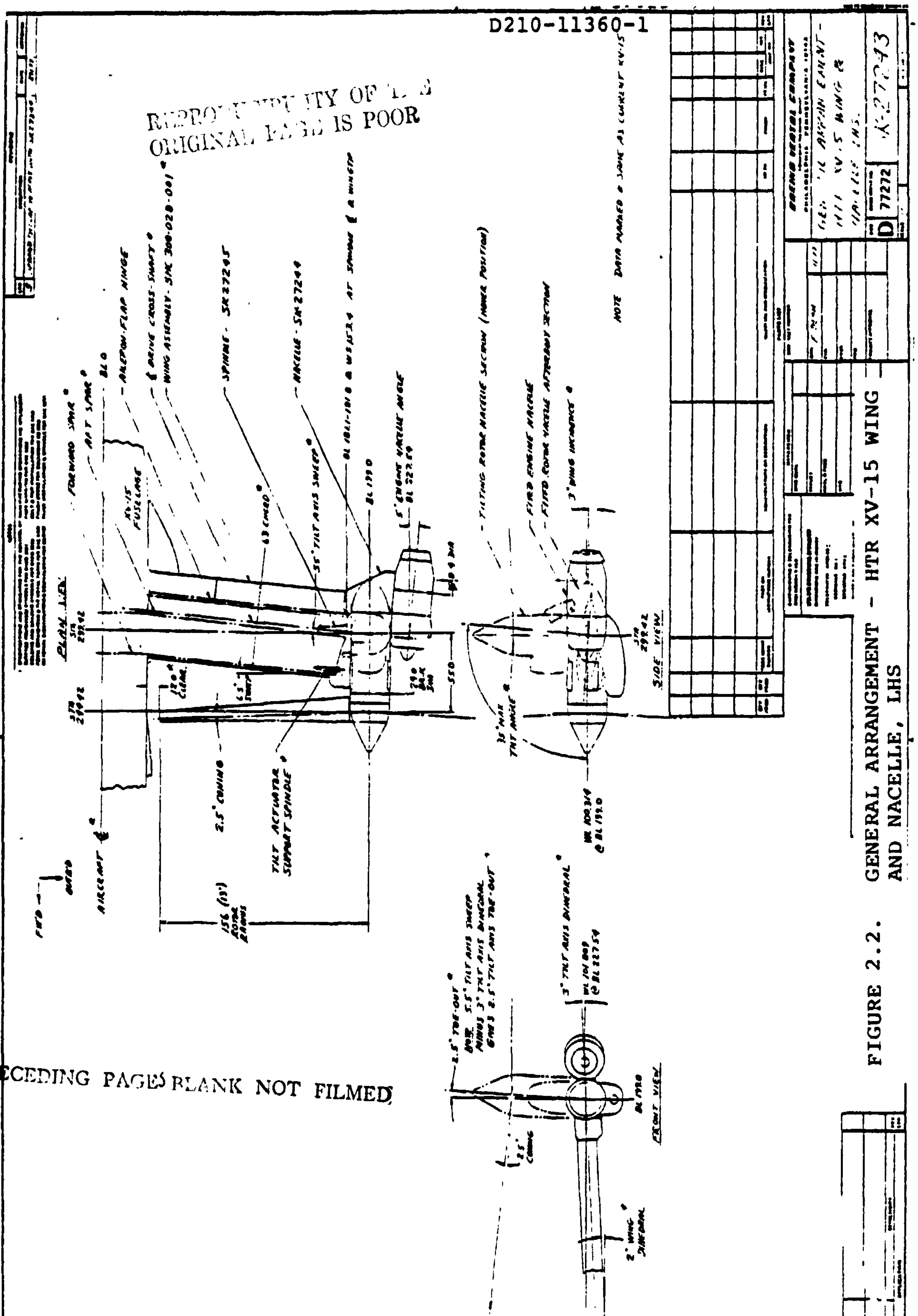


FIGURE 2.2. GENERAL ARRANGEMENT - HTR XV-15 WING AND NACELLE, LHS

current XV-15 arrangement in that a spindle protruding inboard from the nacelle, concentric with and surrounding the drive cross-shaft, is inserted into fittings in the wing mounted off the back of the rear spar. The difference is that while the current XV-15 nacelle spindle is set in sleeve bearings so it can rotate through 95° (here the whole nacelle tilts), the new HTR spindle does not rotate during nacelle tilt because it must keep the engine nacelle and inboard nacelle afterbody fixed. A torque reaction arm portion of nacelle fixed structure, of which the inserted spindle is also a part, is tied to the wingtip closing rib to keep the system from rotating (tilting). The tilting rotor nacelle is supported from this fixed nacelle structure by two sleeve-type trunnion bearings concentric with and surrounding the drive cross-shaft. The cross-shaft through the nacelle is swept forward 5.5° to match up with the XV-15 wing cross-shaft sections just inboard of the inserted nacelle support spindle. The nacelle power drive system consists of a right angle bevel gear transmission connecting the engine to the cross-shafting, an intermediate transmission with a bevel gear set connecting the rotor drive to the cross-shaft in a manner allowing the rotor nacelle to tilt, a rotor transmission consisting of a single stage planetary gear set to reduce speed to rotor RPM, and three short sections of connecting shafting. The intermediate and rotor transmissions are mounted to tilting nacelle structure; the engine transmission is mounted directly to the fixed engine nose case.

The tilting rotor nacelle is comprised of the rotor assembly, including upper controls, rotor support structure, two drive transmission, shafting, accessory drive system, and various subsystem components. Electrical, lube oil, and hydraulic lines must pass across the tilt joint from the fixed afterbody section. It is believed that this can be effected simply by properly guided slack loops in the lines over, under, and alongside the intermediate transmission with no special transfer joints required since the nacelle tilts only 95° maximum. Guides must be provided to keep slack lines from contacting the rotating cross-shaft components and avoid chafing. All rotor loads other than torque are taken through the rotor mounting bearing into the upper cover, and then directly aft (or down in hover flight) through the nacelle structural shell to the aft bulkhead and mounting trunnion bearings, and on to the nacelle fixed structure (and wings). These loads do not feed into the transmissions. The upper cover is bolted in six places to fittings on the structural shell. The rotor nacelle structural diameter is nine inches less than the nacelle basic outside diameter to allow space within contour for the rotor power actautors and their associated hydraulic and electric harnesses outside the structure. These components are readily accessible via non-structural panels in the nacelle fairing skin. The intermediate transmission is simply mounted with four bolts to the aft bulkhead of the structural shell, well away from the rotor load paths. The two tilt nacelle transmissions are connected by a shaft and flexible coupling drive system incorporating torque and speed sensors. The rotor hub is flange-mounted to the main transmission output shaft. The rotor control swashplate assembly is located between the hub and upper cover. Accessories are housed in an easily accessible underslung bay outside the nacelle basic structure and covered with light, removeable fairing skin panels. The accessory drive gearbox (AGB), one per nacelle, is powered whenever rotors are turning, being driven off the intermediate transmission via a short shaft and coupling system. The AGB powers hydraulic pumps, a flight control generator (on one side), various drive system lube pumps, and also continuously drives a blower pulling air through the transmissions cooler and on through the accessory compartment. Oil tanks for both main and engine transmission are also located in easily accessible locations outside nacelle structure. The accessory section, oil tanks, and intermediate transmission oil pump all tilt with the rotor, with the latter slipping out of a cavity in the afterbody fairing as the nacelle tilts up for hover flight.

The fixed afterbody section of the inboard nacelle continues the smooth cruise flight external contours aft to a rear end fairing which is a continuation of the wing trailing edge. This section contains the fixed nacelle structure tying the engine nacelle to the wingtip and providing the bearing supports for the rotor nacelle. It also mounts and houses subsystem components, and incorporates a clamshell fairing door on its upper surface which is driven open by the tilting-up action of the rotor nacelle. This door is hinged on the nacelle structure and spring preloaded against the rotor nacelle so the total body will be smoothly faired in cruise flight and remain against the nacelle surface in hover flight. The section of the nacelle aft of the door is made up of removeable fairings so that quick access to components within may be gained. These items include hydraulic modules, valves, and accumulators, engine fire extinguishant bottles, and in one nacelle bleed air line components. All powerplant service lines, including main electrical harness and fuel line, run from the wingtip across this fixed afterbody section to the engine compartment.

The engine nacelle is mounted directly to and outboard of the above fixed afterbody structure. It contains the engine, powerplant subsystem items including the engine oil tank, cowling, and engine transmission. The transmission is mounted on the engine in a faired bullet centerbody within the engine air inlet annulus. The engine inlet is designed as a compromise between low and high speed flight conditions as is the tail pipe. The outer forward annulus of the cowl is mounted on the engine; the remainder of the cowl aft is tied to fixed afterbody structure. The engine can be removed upward via a large removeable access panel. No other part of the nacelle is disturbed during an engine removal replacement. A second large servicing access panel is located on the bottom of the cowl. The engine is the identical model to that on the current XV-15 aircraft. The same electrical generator is mounted as an accessory on the engine pad. In the HTR XV-15 the engine is controlled as part of the electric fly-by-wire system.

2.2 AIRFRAME STRUCTURE

2.2.1 Structural Arrangement

The structural arrangement of the HTR XV-15 nacelle is shown in Drawings SK-27245 (Figure 2.3), and SK-27245-1 and -2 (Figures 2.4 and 2.5) pertaining respectively to the total assembly, tilting rotor nacelle structural shell, and structural adapter in the fixed nacelle afterbody. Drawing SK-27245 (Figure 2.3) shows a plan view, with nacelle in cruise position, of the major structural elements including the current XV-15 wing and fixed and tilting nacelle portions. The arrangement was dictated by four primary factors:

- The desire to minimize required modifications to the current wing and resulting cost.

- The decision to use a fixed engine.

- Integration with the selected drive system arrangement, and separation of rotor load paths from gearbox cases.

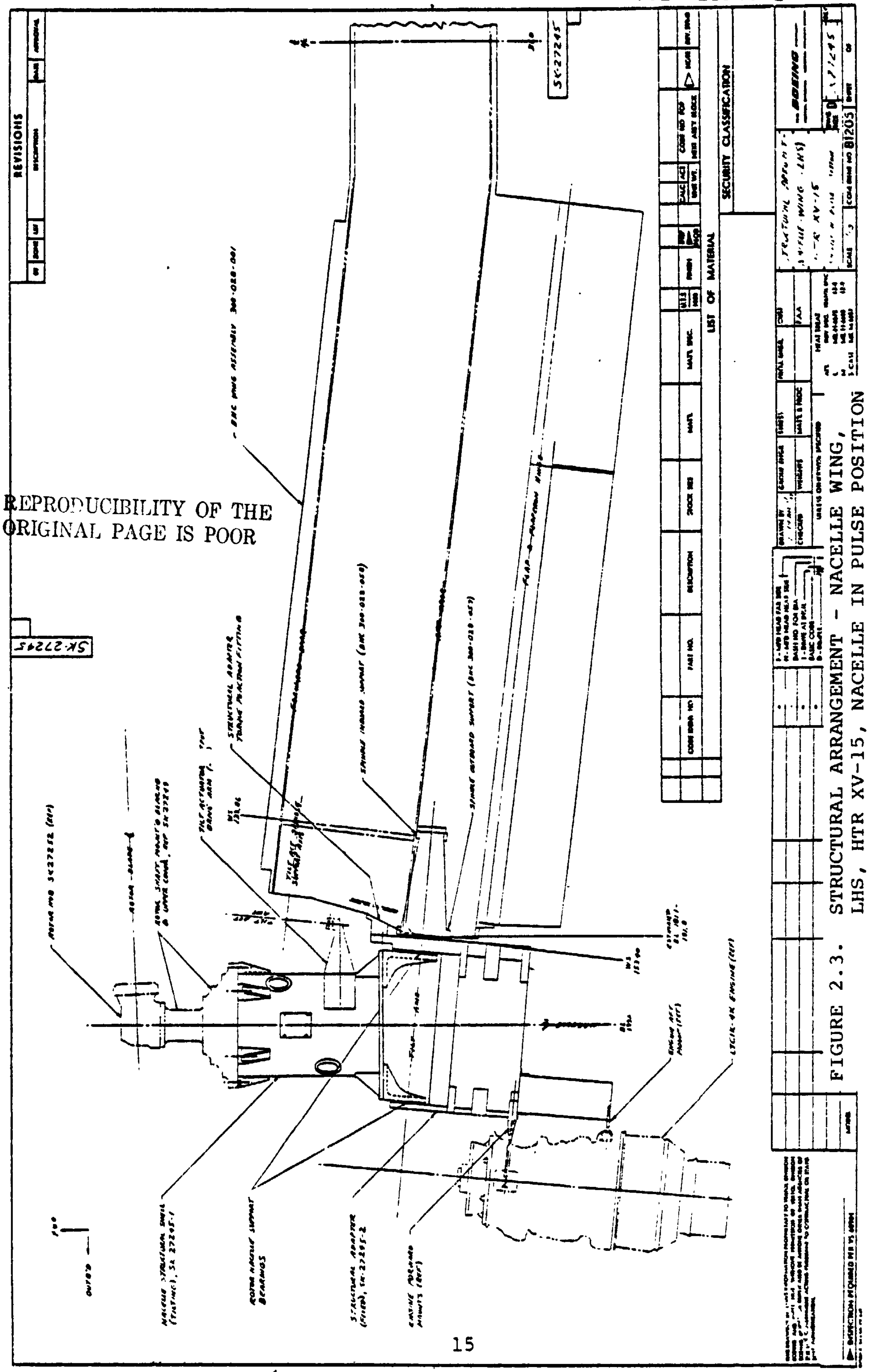


FIGURE 2.3. STRUCTURAL ARRANGEMENT - NACELLE WING, LHS, HTR XV-15, NACELLE IN PULSE POSITION

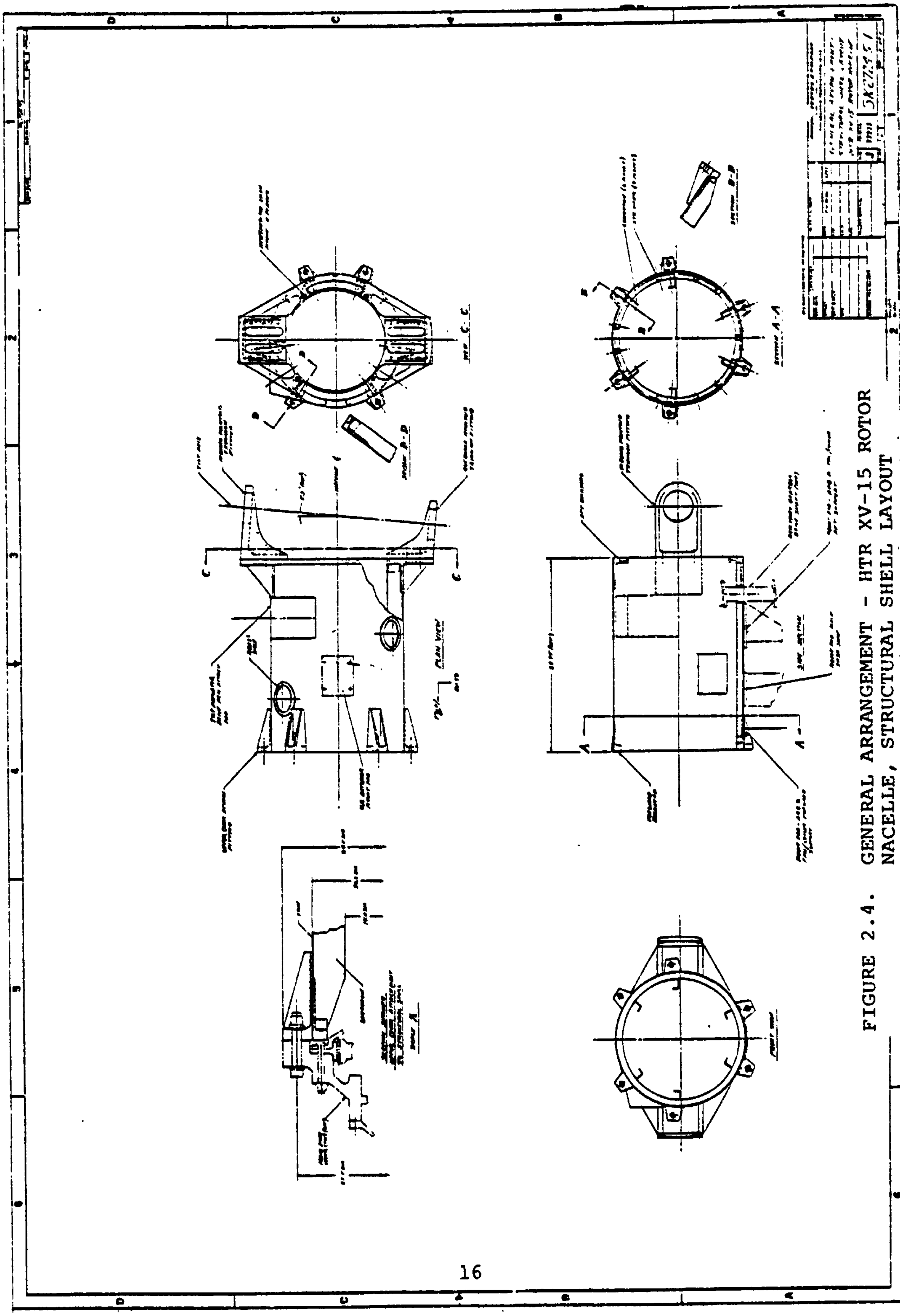


FIGURE 2.4. GENERAL ARRANGEMENT - HTR XV-15 ROTOR NACELLE, STRUCTURAL SHELL LAYOUT

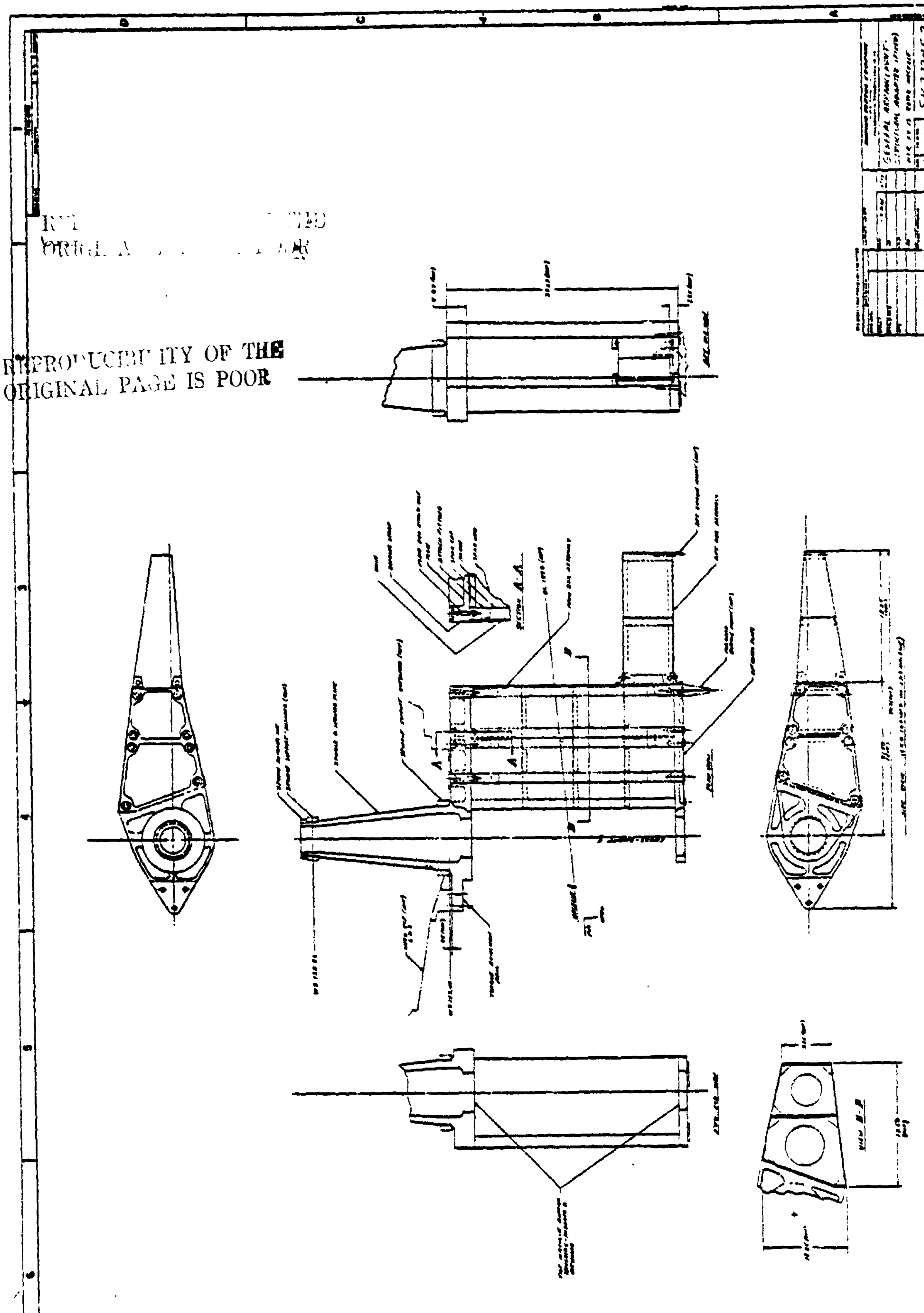


FIGURE 2.5. GENERAL ARRANGEMENT - HTR XV-15 ROTOR NACELLE, STRUCTURAL ADAPTER (FIXED)

- Provision for integrity of nacelle structure along with accessability of components.

2.2.2 Rotor Nacelle Structure - (Drawing SK27245-1, Figure 2.4)

The tilting rotor nacelle structure supports the rotor, two gearboxes, controls and accessories. The shell is a semi-monocoque skin/longeron/stringer assembly tied in to stiffening bulkheads at each end. Attach fittings are provided forward for the main transmission upper cover through which rotor loads are taken. Forged aluminum angle trunnion fittings aft pick up the inboard and outboard nacelle mounting bearings. The brackets are separate bolted-on elements to aid in nacelle disassembly and in unhanding these assemblies. Pads are provided for the tilt actautor output drive, rotor control actuators, main transmission oil sump, and accessory drive items. Cut-outs are required for access to interior drive shaft couplings and for passage of the accessory gearbox drive shaft. Boxed-in backups are provided for the two aft bulkhead ears which support the angle trunnions.

With the exception of the output arm, the current XV-15 tilt actuator system, including wing structural support and nacelle down-stop, is planned for use. Thus no wing changes should be required in this area. To confirm this, however, more detail drawings of the XV-15 wingtip area are needed.

Although Drawing SK-27245-1 (Figure 2.4) shows the general concept of the nacelle structure, it requires updating to agree with the general arrangement of Figure 2.2 as follows:

a) An increase in length of one inch from 28.75 to 29.75 inches.

b) Modified arrangement and location around the periphery of upper cover attach fittings to agree with main rotor transmission Drawing SK-27249 and a resulting rearrangement of shell longerons and stringers to match fitting locations.

c) Change in cut-out location for accessory drive gearbox shaft to match SK-27251 AGB. Revision in mounting pads for AGB. Review of access hole locations.

d) Addition of intermediate reinforcing rings near tilt actuator attach and rotor boost actuator mount pads.

2.2.3 <u>Fixed Structural Adapter</u> (Drawing SK27245-2, Figure 2.5) The nacelle structure is fixed to the wing by a spindle protruding into wingtip supports just aft of the rear spar, like the XV-15, and by a torque reaction fitting (unlike the current XV-15, the aft part of this nacelle is non-tilting). This concept calls for the current wing rib (spindle) supports to be used with the support bearings removed; thus the new spindle O.D. can be about 10-25% greater than the equivalent

XV-15 item. Nacelle loads are taken into the wing through these two supports except adapter torque restraint is provided by a fitting tied to the wingtip structure. Currently there is insufficient wing structural design data to determine feasibility of the torque restraint shown.

The fixed aft part of the nacelle structure (adapter) supports the engine, engine-supported items (including cowling and engine gearbox), aft nacelle framing and skin, subsystem components and lines, and the tilting rotor nacelle via two support bearings (plain sleeve type) at the tilt axis.

The spindle and inboard plate have been combined into one forging and the basic plate thickness increased for more stiffness. This portion would be an aluminum forging along with the outboard plate, these being bolt connected by a built-up main box assembly having lateral spars and longitudinal ribs. The forward ends of the plates provide trunnion housings for the rotor nacelle support bearings. A smaller box assembly is bolted to the main box to support the aft engine mount.

2.3 DYNAMIC SYSTEMS

2.3.1 Rotor

The rotor is a three blade unit of 26-foot diameter utilizing hingeless fiberglass blades of 18.85 inch chord. The design characteristics have been selected to provide good performance throughout the flight regime from hover to cruise flight. The rotor system consists of blades, hub assembly, spinner, and upper controls from the pitch arm back to the actuator input side of the swashplate. The blade and cuff assembly are shown on Drawing SK-27253. Drawing SK-27252 depicts the rotor hub assembly, and the upper controls located between the hub and main transmission are presented in Drawing SK-27255 (Fig. 2.6 and Fig. 2.7). The spinner and overall arrangement of the rotor is shown in Drawing SK-27244. The rotor is characterized by a one-piece basic hub and a two pin blade retention connecting pitch shaft and blade root just ouboard of the hub. No flapping or lead-lag hinges are present - the inboard section of the blade shank provides the required flexibilities of the rotor. The rotor hub is designed with a minimum of parts to promote reliability and ease maintenance problems.

2.3.1.1 Blade and Cuff

The design data for the blade and cuff were produced under the IR&D program, and are considered proprietary to The Boeing Company. These data are provided in Appendix II.

2.3.1.2 Hub

The hub design to accommodate the twin pin blade retention is considered proprietary to The Boeing Company, and was performed under the IR&D program. This data is provided in Appendix II.

2.3.1.3 Spinner

The rotor spinner provides an aerodynamic fairing for the rotor under the higher speed conditions of cruise flight. It consists of a conical forward section, a drum-type afterbody section matching with the contour of the non-rotating nacelle fairing, and risers and fill-in panels for each rotor blade.

The spinner is supported by the hub. Forward and aft spinner bulkheads are mounted on front and back faces of the hub with the spinner forward section bolted to the outer periphery of the forward bulkhead. The conical forward section is constructed of fiberglass to integrate with the installation of the instrumentation telemetry transmitter system wherein a transmitter is mounted on the rotor hub front face. The spinner afterbody has cut-outs for each blade shank and employs small built-up riser or 'island' sections matching with the forward contours of the blade cuff inboard station and driven through the blade angle range as required by the rectangular section of the local blade shank. Removeable fill-in panels behind each blade hole complete the spinner assembly. The bulkhead and afterbody sections of the spinner are constructed of aluminum alloy.

2.3.1.4 Upper Controls

The preliminary design of the upper controls is shown in Sheets 1 and 2 of SK-27255 (Figures 2.6 and 2.7). The swashplate is of conventional design and is supported by four scissors on the stationary side which center the assembly as well as reacting the stationary ring torque. The rotating side is driven by two scissors providing redundancy as well as balance.

In this instance redundancy costs no additional weight since balance weights would be needed anyway.

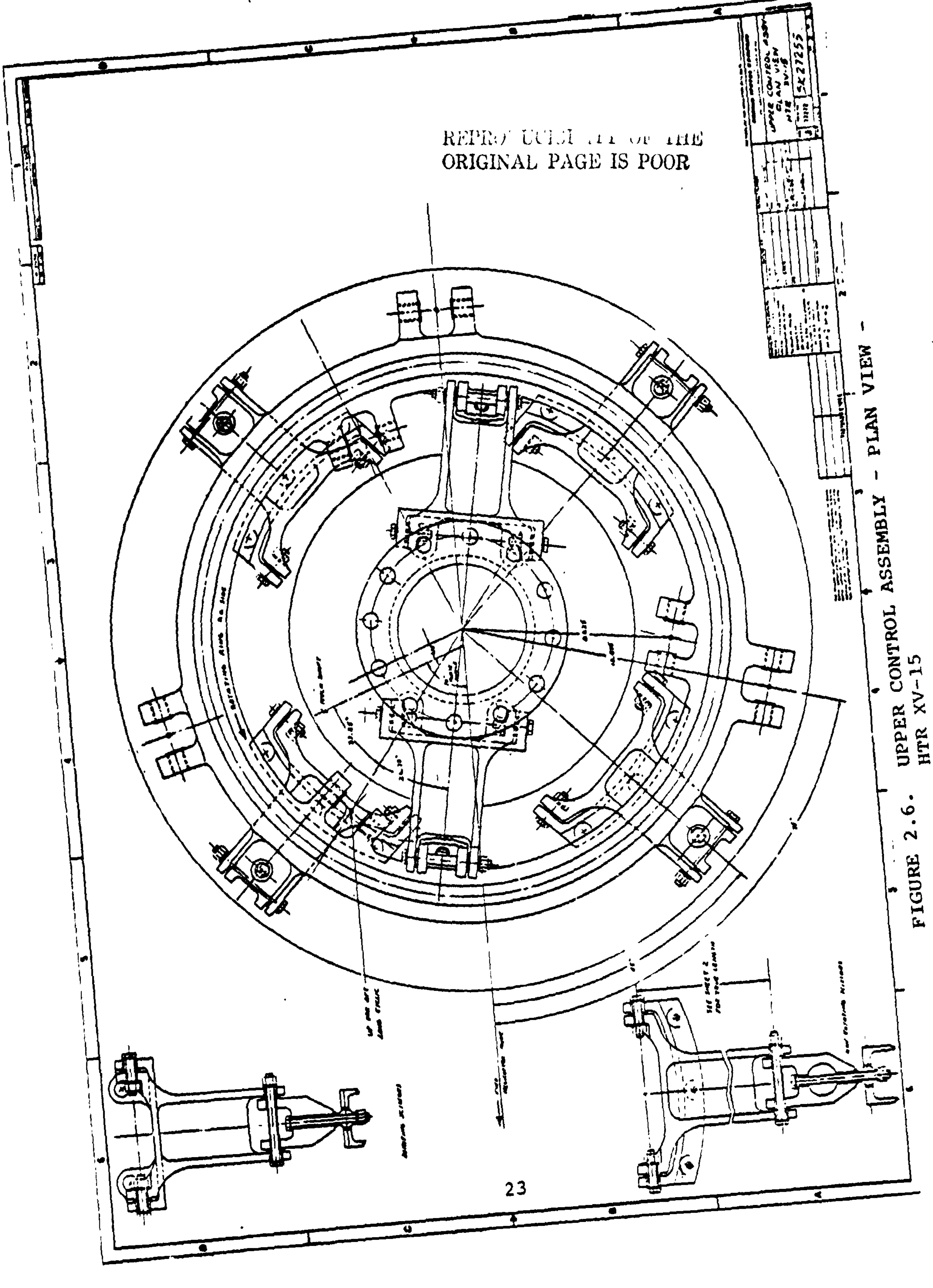


FIGURE 2.6. UPPER CONTROL ASSEMBLY - PLAN VIEW - HTR XV-15

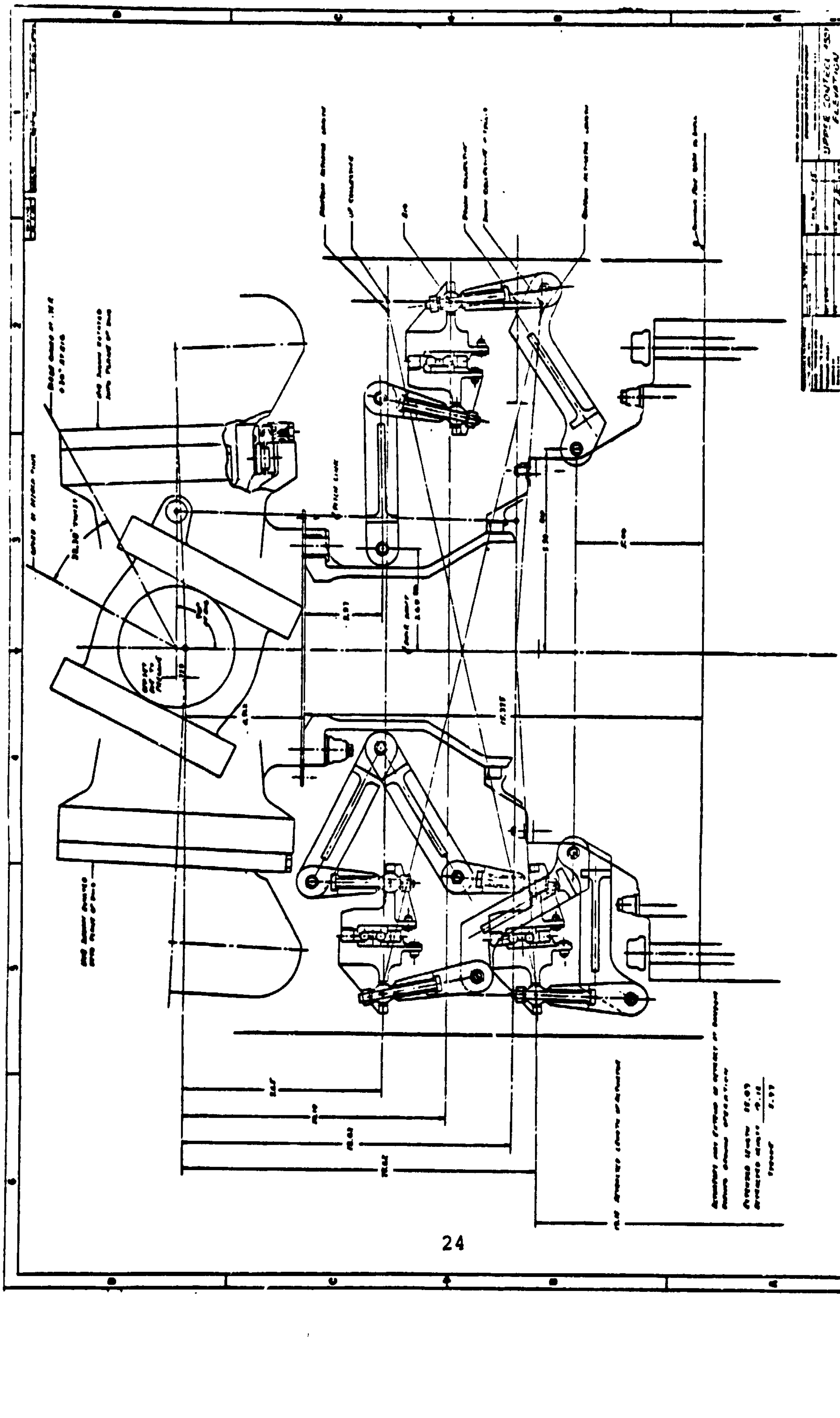


FIGURE 2.7. UPPER CONTROL ASSEMBLY - ELEVATION - HTR XV-15

This type of arrangement is essentially the same as the YUH-61A and can accommodate the control motions necessary for the HTR XV-15 as shown in Figure 2.7.

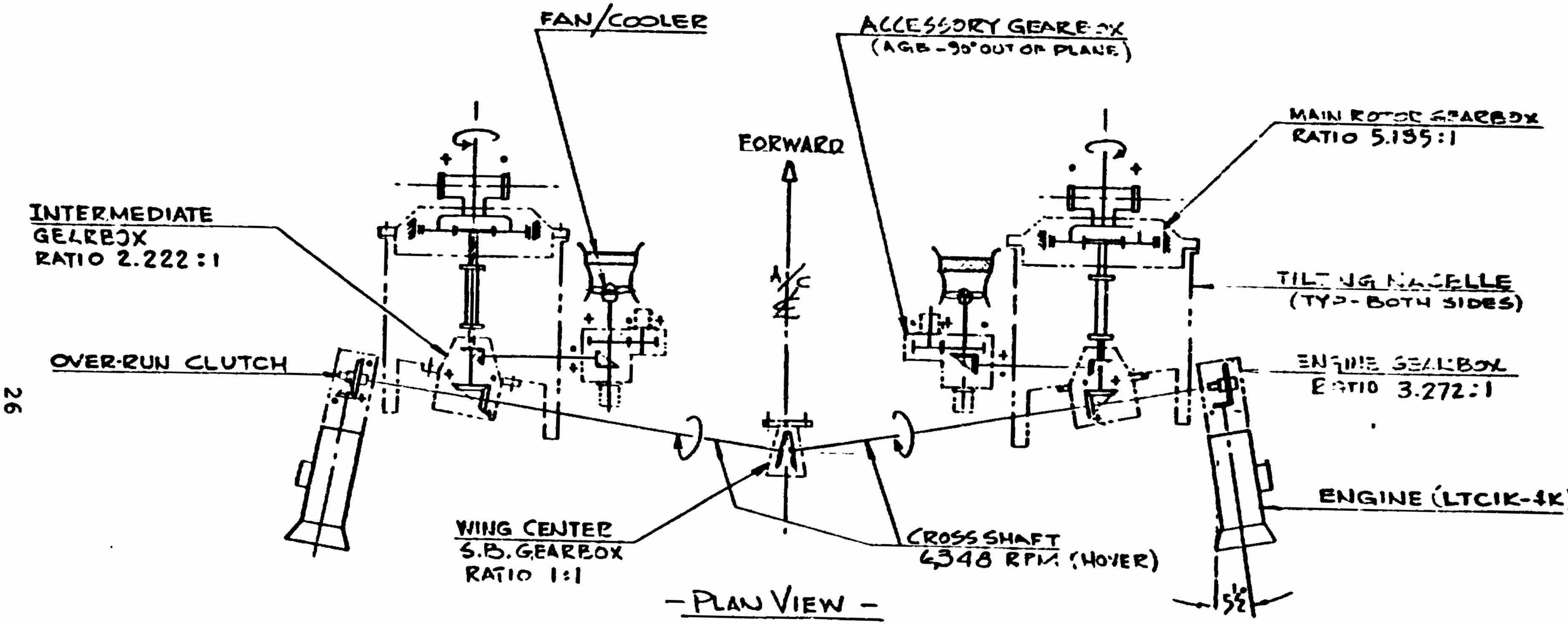
2.3.2 Drive Train

2.3.2.1 System Description

The HTR XV-15 drive system connects two Lycoming LTC1K-4K engines to a Boeing rotor system and interfaces the existing Bell cross shaft drive inboard of the nacelles. A schematic of the system is shown in Figure 2.8. A primary design objective of the drive system has been to maintain the simplest and most efficient arrangement consistent with the nacelle and wing configuration. The Boeing drive system is designed to current state-of-the-art criteria and allowables, and thus represents a minimum-risk approach.

Arrangement (SK-27244, Figures 2.2 and 2.8) - Forward of each engine, a right-angle bevel gearbox transmits power to the rotor system and cross shaft. The output gear drives the rotor system through an overrunning clutch, allowing for engine-out operation. The engine bevel gearbox is connected to the intermediate box through a tubular drive shaft and flexible couplings. The ratio of the engine box is set to maintain the existing XV-15 maximum cross shaft rpm of 6392. A gear ratio of 3.27 results.

The intermediate gearbox incorporates a bevel mesh to direct power to the rotor as well as a through-shaft to connect to the wing cross shaft. The bevel gear reduction ratio is set to take maximum advantage of a single stage planetary reduction



– PLAN VIEW –

DRIVE SYSTEM SCHEMATIC

RPM:	HOVER	CRUISE	RATIOS
ROTOR	551	386	
MAIN GEARBOX	2,857	2,001	5.185:1
INTER – FAN	6,914	4,843	1:2.420
INTERMEDIATE	6,348	4,447	2.222:1
ENGINE GEARBOX	20,771	14,551	3.272:1

FIGURE 2.8. DRIVE SYSTEM SCHEMATIC

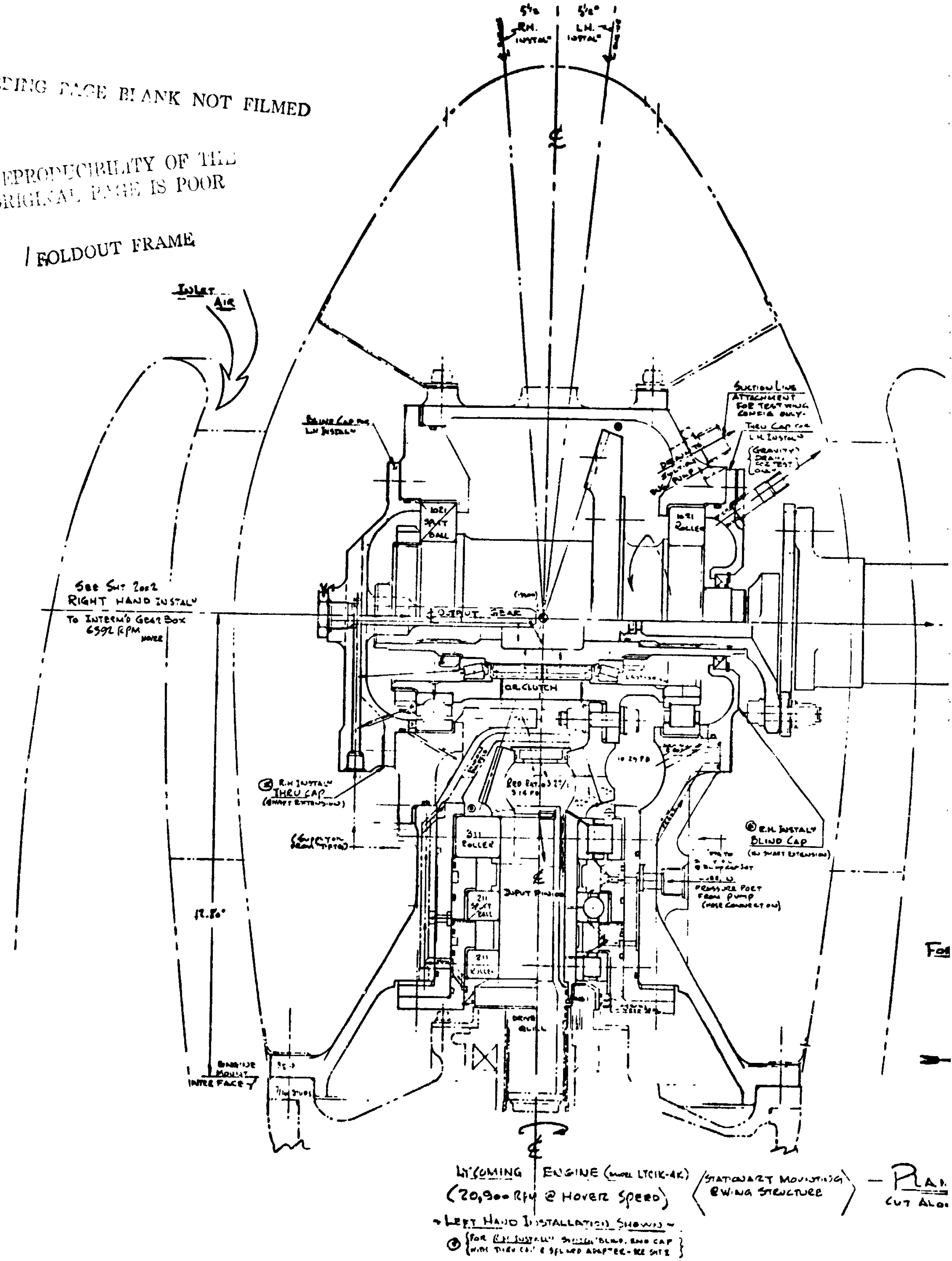
in the rotor gearbox. The resultant ratio is 2.22 to 1.

The rotor box uses the same ratio (5.18) and number of planetaries as the highly successful YUH-61A planetary, and is in effect a scaled-down version of this planetary system on which over 4,400 test hours have been accumulated. The forward end of the rotor box comprises a rotor shaft and hub mounting flange and a cover assembly to react rotor loads to the nacelle structure. The direct load path provided by the cover eliminates rotor loads and consequent distortions from the gear train. Further isolation of rotor loads is provided by the flexible coupling drive shaft between the rotor and intermediate boxes.

Aircraft accessories, lubrication pumps and oil cooler blower are driven by an accessory gearbox mounted below, and driven from, an extension of the intermediate box.

2.3.2.2 Gearboxes

Engine Box (SK-27246, Figures 2.9 and 2.10) - The engine box is designed to a continuous duty rating of OEI (one engine inoperative) power of 1710. The AEO power input is 1468. There is thus a 14% margin between design power and normal condition maximum power, providing a reserve capacity for additional reliability at a small increase in weight.


R.H. INSTAL
L.H. INSTAL
INLET AIR
BLIND CAP FOR LH INSTAL
SUCTION LINE ATTACHMENT FOR TEST WING CONFIG ONLY
THRU CAP FOR L.H. INSTAL
1021 SPLIT BALL
1021 ROLLER
SEE SHT 2 OF 2
RIGHT HAND INSTAL
TO INTERM'D GEAR BOX
6592 RPM HOVER
OUTPUT GEAR
OR. CLUTCH
R.H. INSTAL THRU CAP (SHAFT EXTENSION)
R.H. INSTAL BLIND CAP
311 ROLLER
211 SPLIT BALL
211 ROLLER
INPUT PINION
PRESSURE PORT FROM PUMP (HOSE CONNECTION)
12.80"
ENGINE MOUNT INTERFACE
DRIVE QUILL
LYCOMING ENGINE (MODEL LTC1K-4K)
(20,900 RPM @ HOVER SPEED)
STATIONARY MOUNTING & WING STRUCTURE
LEFT HAND INSTALLATION SHOWN

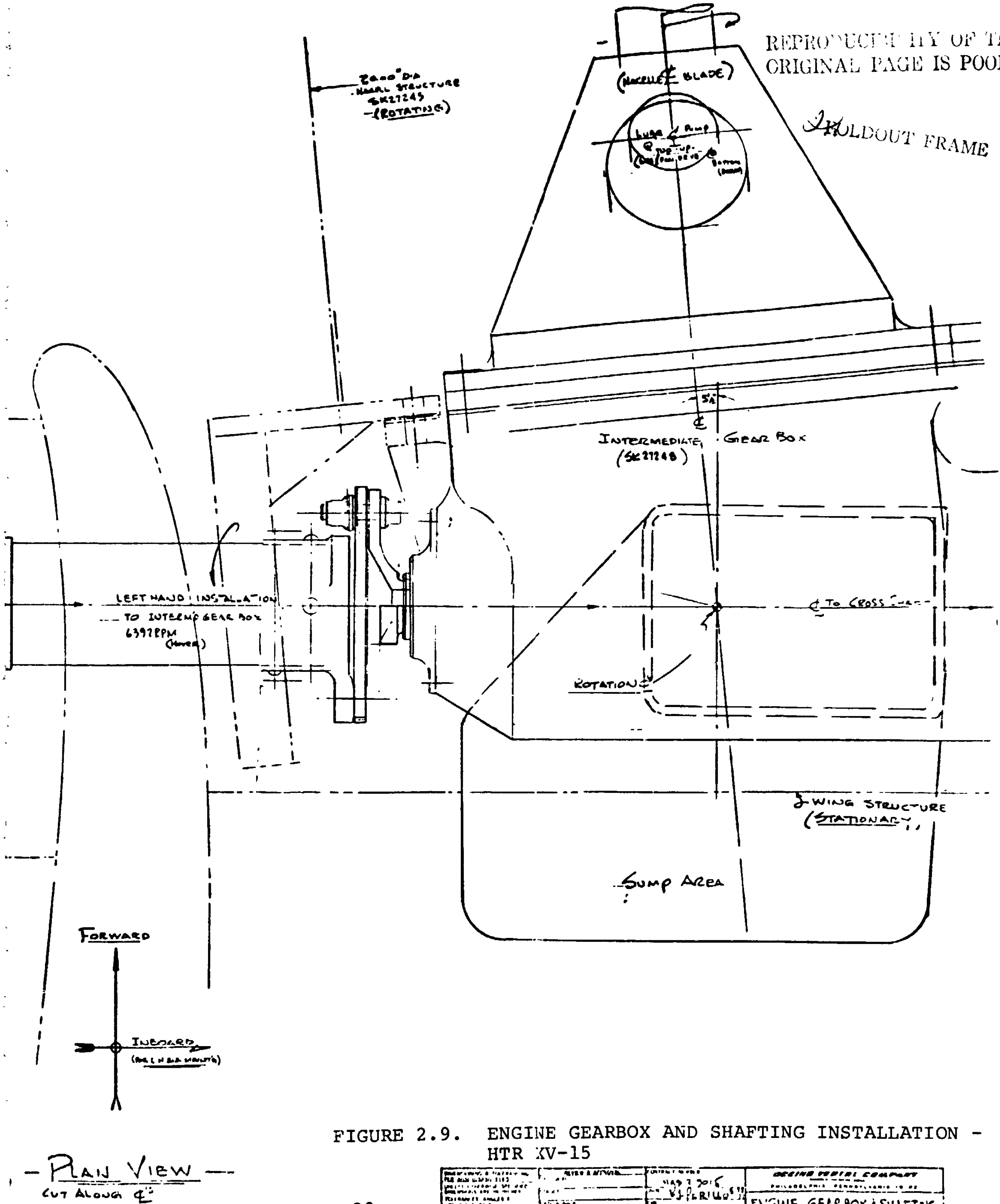


FIGURE 2.9. ENGINE GEARBOX AND SHAFTING INSTALLATION - HTR XV-15

BOEING VERTOL COMPANY	
ENGINE GEAR BOX & SHAFTING INSTAL'N - HTR XV-15	
77272	SK27246

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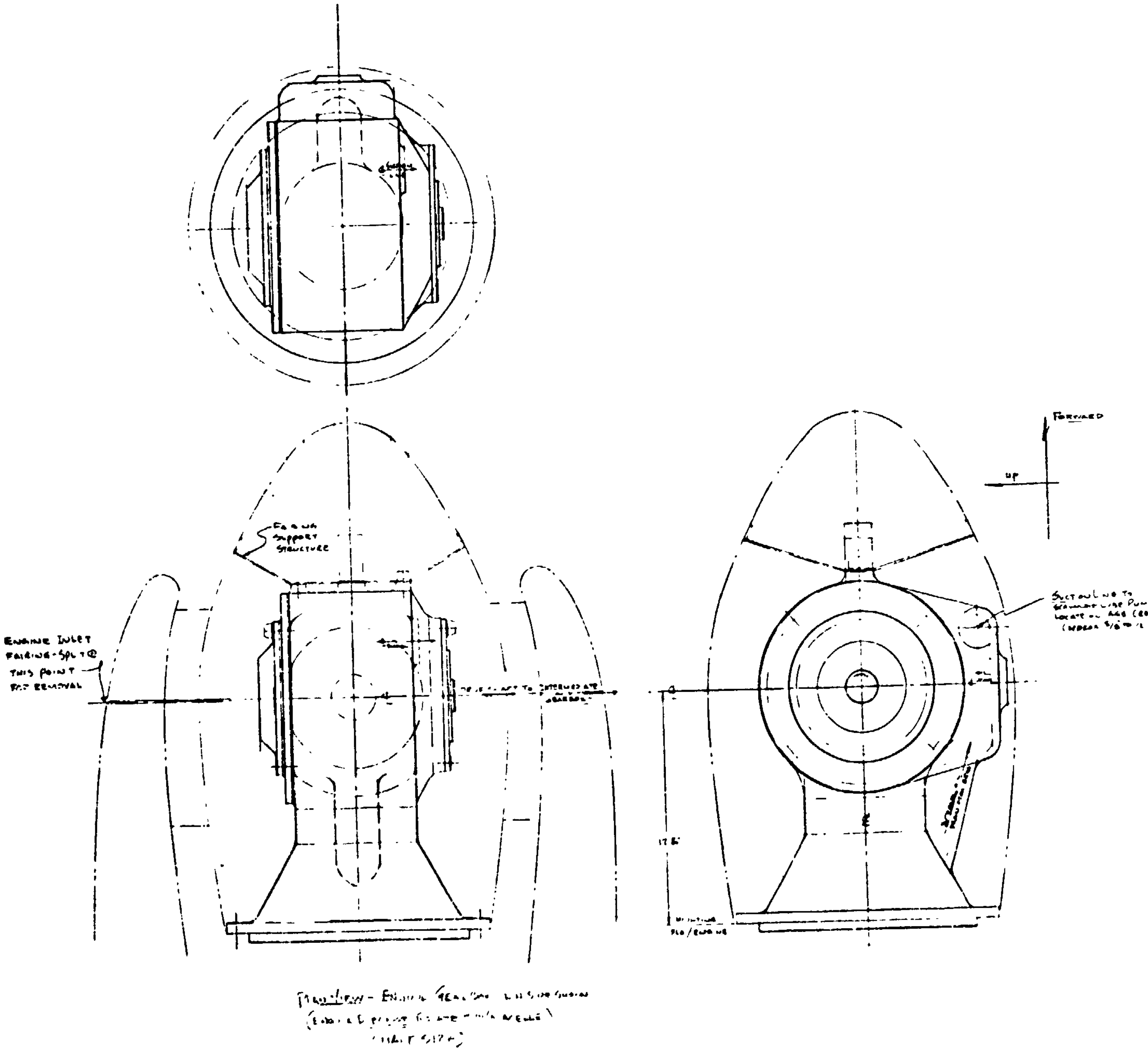

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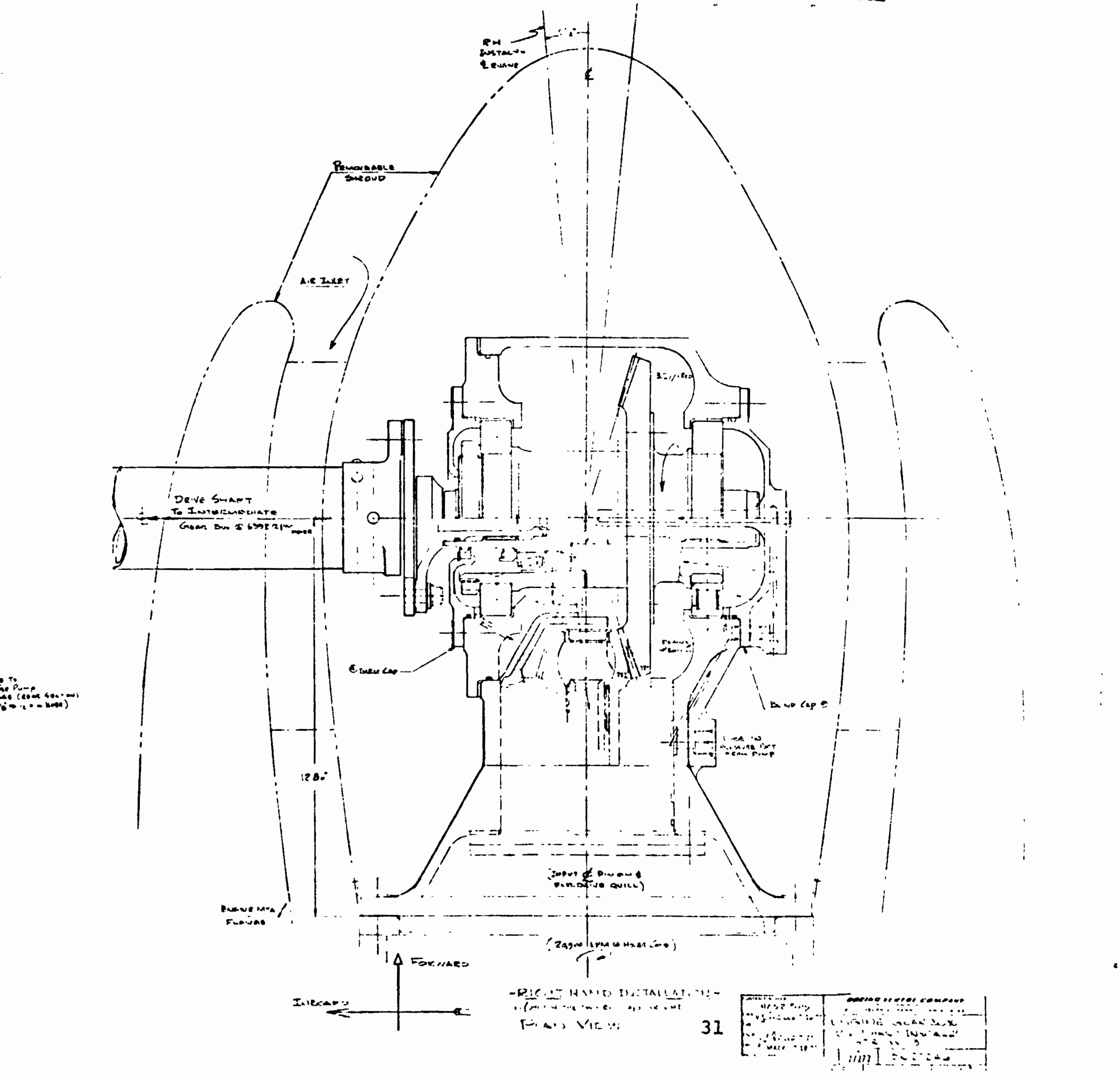

FIGURE 2.10. ENGINE GEARBOX - RIGHTHAND INSTALLATION - HTR XV-15

The engine box supports and contains a right angle spiral bevel gear set with a ratio of 3.272 to 1. This set is the highest speed mesh in the power train, but due to the moderate power requirements, it operates within conventional design experience. For example. Pitch line velocity of 16,500 fpm compares to experience with the CH-47C of 30,000 fpm; bearing velocity of 1.5 million compares to a DN of 1.7 million.

Engine box gearing is summarized in Table 2.1. Gear material is a carburizing grade CEVM (comsumable-electrode vacuum melt) steel sucn as AISI 9310 or VASCO X-2. Gears are carburized and ground. Axial shiming provision in the housing provide for control of the gear load pattern in the assembly. Bearings are ABEC Class 5 precision bearings. They will be individually designed to provide optimum strength and capacity for their intended usage. A summary of bearing sizes and calculated lives is shown in Table 2.2. Each bearing is jet lubricated from passages incorporated into the housing. Bearings are supported in the housing by steel liners.

The gearbox housing is designed as a lightweight, high strength magnesium casting of ZE41A alloy. Detail studies to be made will compare costs and weight of castings to machined magnesium billets. In experimental, small-lot fabrication the lead time and cost of completely machined housings may be less than castings.

TRANSMISSION	LOCATION	TEETH	RATIO	P.A. ϕ	S.A. ψ	Pd	FACE (INCH)
ENGINE	PINION GEAR	22 72	3.272	20°	25°	7.000	1.61
INTERMEDIATE	PINION GEAR	27 60	2.222	20°	26°	5.328	1.82
ROTOR	SUN PLANET RING	27 43 113	5.185	25°	N/A	7.062	1.75

GEAR DESIGN CRITERIA:

	Spiral Bevels	Spurs
$S_b \Leftarrow$	37,000 psi	$\Leftarrow$ 37,000 psi
$S_c \Leftarrow$	235,000 psi	$\Leftarrow$ 165,000 psi
$T_f \Leftarrow$	500°F	$\Leftarrow$ 300°F

TABLE 2.1. HTR XV-15 GEAR SUMMARY

HTR XV-15 BEARING SUMMARY

(See Figure 2.11)

TRANSMISSION	LOCATION (SEE SCHEMATIC)	LOAD (LBS)	SPEED(1) (RPM)	TYPE	SIZE	LIFE (2)(3) (HRS)
ENGINE	A	3876	18,000	ROLLER	311	2,580
	B	1347	18,000	ROLLER	211	24,600
	C	1462(T)	18,000	25° BALL	211	2,280
	D	2074	5,500	ROLLER	1021	25,000
	E	933 + 655(T)	5,500	25° BALL	1021	20,000
INTERMEDIATE	A	7819 + 3234(T)	5,500	TAPER	HM 617049	2,627
	B	3123	5,500	TAPER	27620	5,777
	C	8200 + 1172(T)	2,466	TAPER	L 730610	3,162
	D	2225	2,466	TAPER	L 730610	> 5,000
	E	<100	5,500	BALL	210	> 5,000
ROTOR	A	6668	1,251	SPHERICAL	22217	12,000
	B	<100	2,466	BALL	1916	> 5,000
	C	15990	474	TAPER	Special	21,000
	D	14350	474	TAPER	Special	4,580

(1) LOAD & SPEED FROM CUBIC MEAN ANALYSIS OF MISSION SPECTRUM

(2) USING LIFE IMPROVEMENT FACTORS OF 6 FOR M-50 BALL AND ROLLER, 4 FOR TAPERED ROLLER, 3 FOR PLANETARY BEARINGS

(3) LIFE CALCULATED BY CATALOG METHODS

TABLE 2.2. HTR XV-15 BEARING SUMMARY

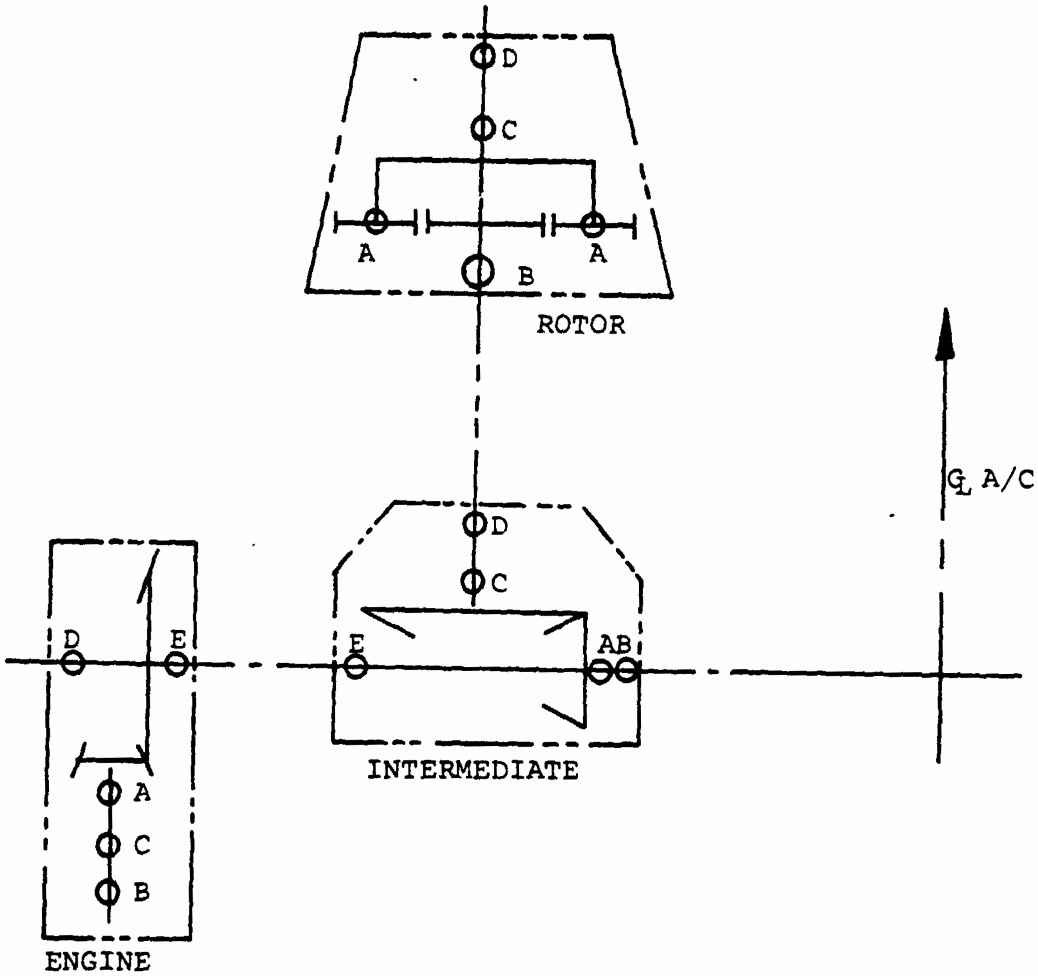

FIGURE 2.11. BEARING SCHEMATIC - (REFERENCE SUMMARY TABLE 2.2)

An overrunning clutch of the sprag type is mounted between the output gear and shaft. In engine-out operation this device disconnects the engine and gearbox from the rest of the power train. Clutch design and sizing parameters are conventional; the clutch as shown is the same as the YUH-61A engine gearbox unit. Clutch lubrication during overrun is provided by an axial oil jet cantilevered from the blank end cap. Oil is directed to the shaft I.D. and is then propelled by centrifugal force to the rubbing surfaces.

The engine gearbox is transferable from left to right nacelle by interchanging output shaft caps, with no internal changes required. Direction of rotation of all elements remains unchanged. Transferability required a 90° angle gear set, as shown in the design. Engineering and fabrication costs are minimized by this feature.

The interface to the LTC1K-4K engine allows the use of a standard, unmodified engine driving through a conventional splined quill shaft. Engine and transmission oil systems are entirely separated. Design study conducted during this program investigated the possibility of sinking the gearbox former inside the engine, thus providing a desirable shortening of the nacelle package. Modification to the standard engine would have been required to accomplish this, however, and it was determined that the extra costs involved did not justify the several inch advantage in length.

Intermediate box (SK-27248), Figures 2.12 and 2.13 incorporates a spiral bevel gear set of 95.5° included angle to transfer cross-shaft power to the rotor gearbox. The design power for this mesh is equal to the maximum per-rotor power of 1468. The ratio change is 2.222 to 1. In addition to transferring and redirecting rotor power, this box also drives the accessory gearbox and an integral lube oil pump through a second, smaller, set of bevel gears.

The particular design conditions of shaft angle and ratio dictated that the pinion as well as the gear be supported in an overhung bearing mounting. The cross-shaft connection is made through a splined quill shaft from one gearbox input to the pinion bore. Both pinion and gear are supported in tapered roller bearings, providing maximum stiffness to the benefit of the two members. The rib velocity of the highest speed tapered bearing is 7100 fpm. The accepted division between normal and high-speed tapered bearings is 8000 fpm. (High speed bearings have been operated successfully at 24,000 fpm in joint Timken-Boeing Vertol tests for the U.S. Army). At 7100 fpm the special lubrication and internal geometry required for high-speed service is not necessary, and conventional jet lubrication is sufficient.

Transfer of this box between left and right nacelles requires two simple reorientations of components. First, the sump and cover plate are interchanged as the gear housing is rolled 180°. Second, the upper cover is detached and reoriented 180°

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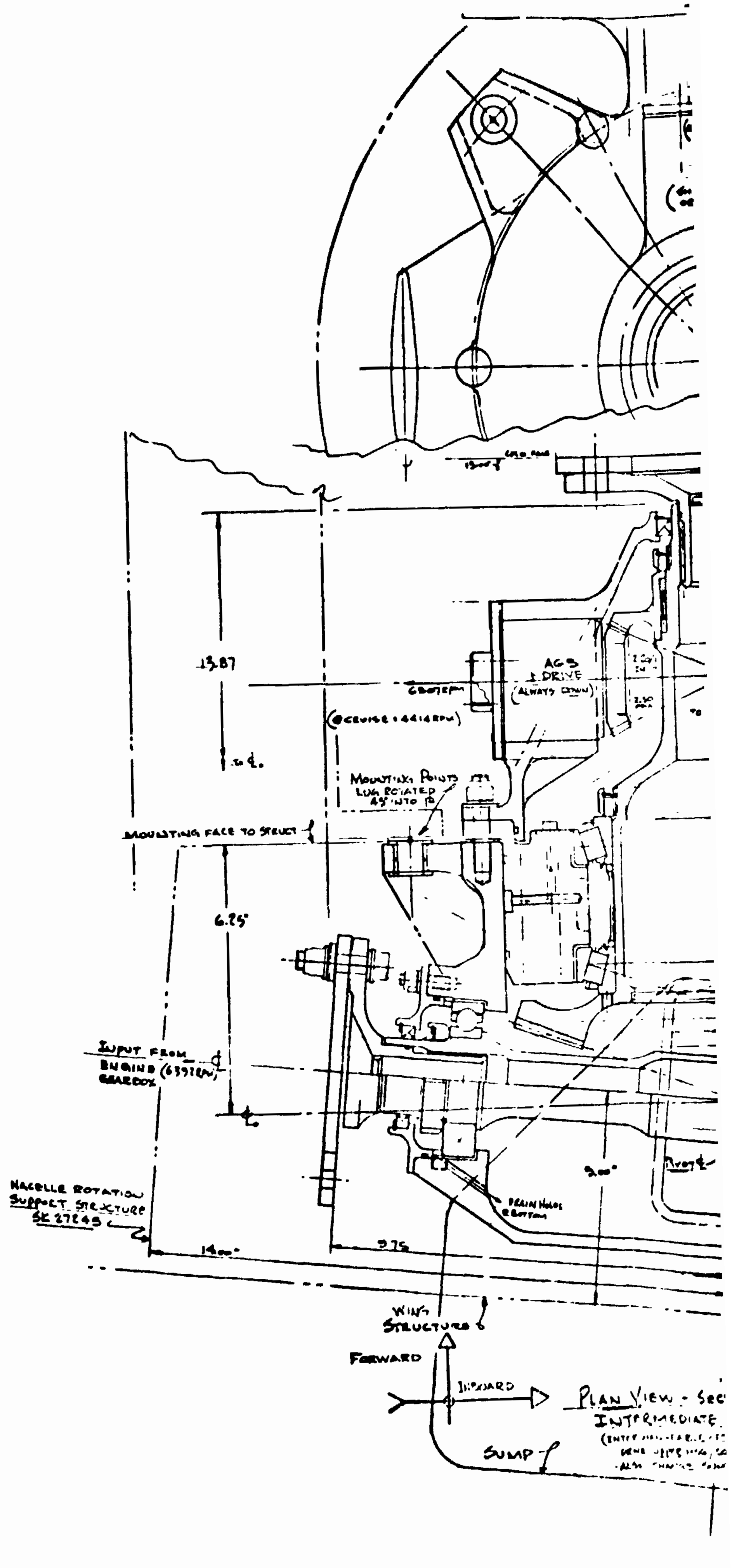

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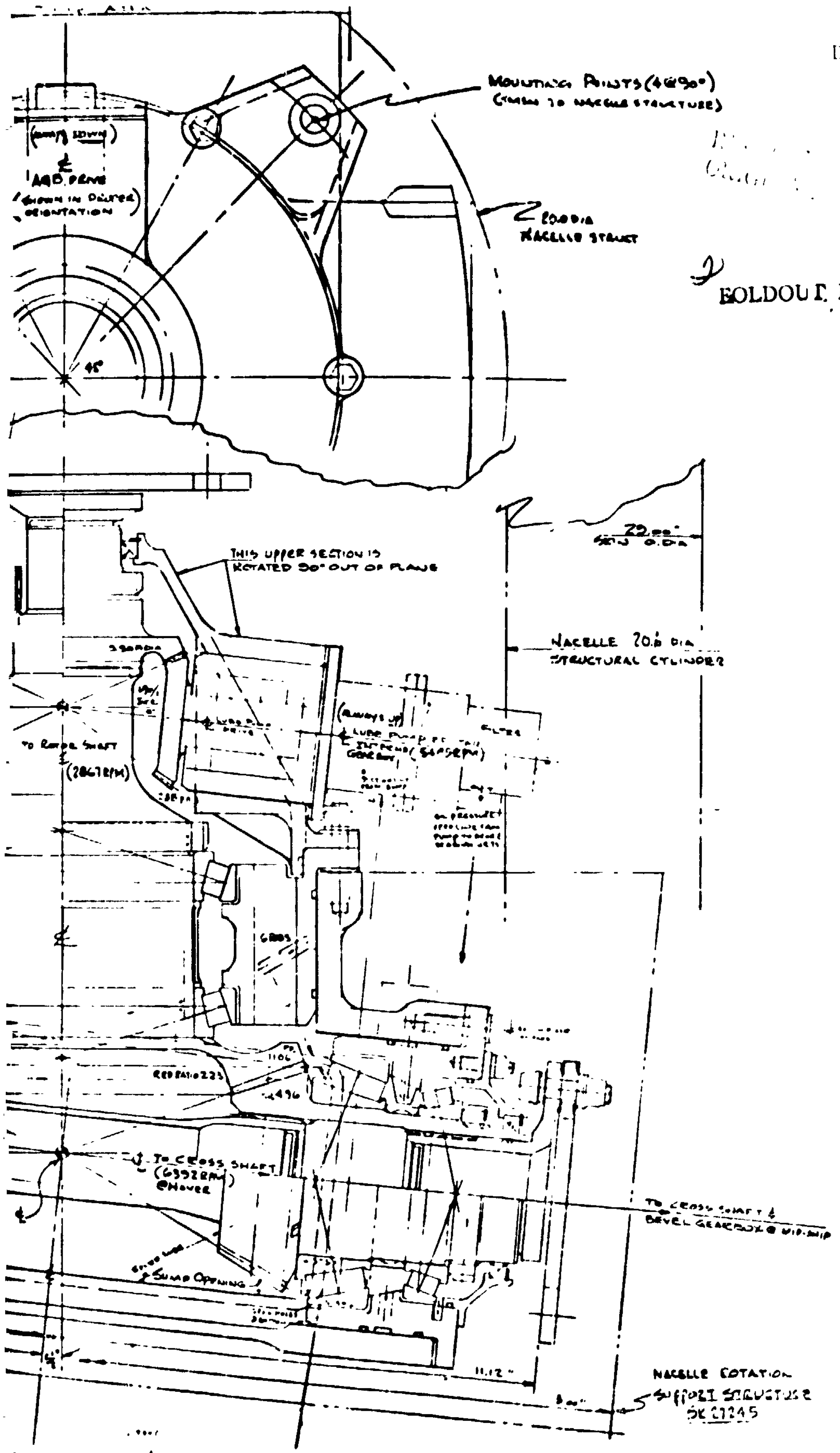

FIGURE 2.12. INTERMEDIATE GEARBOX - INTERNAL ARRANGEMENT - HTR XV-15

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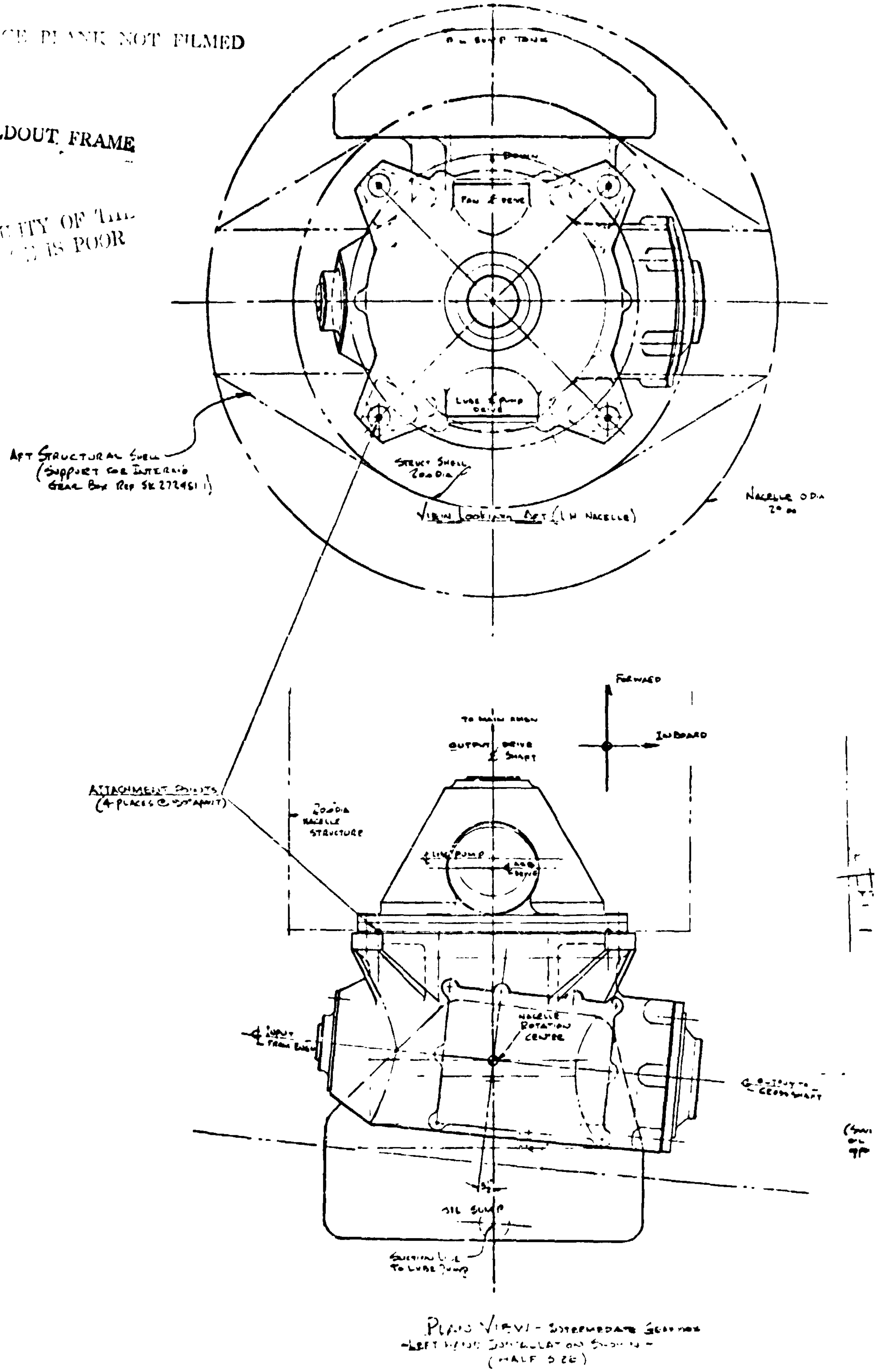

Plan View - Intermediate Gearbox
-Left Hand Installation Shown-
(Half Size)

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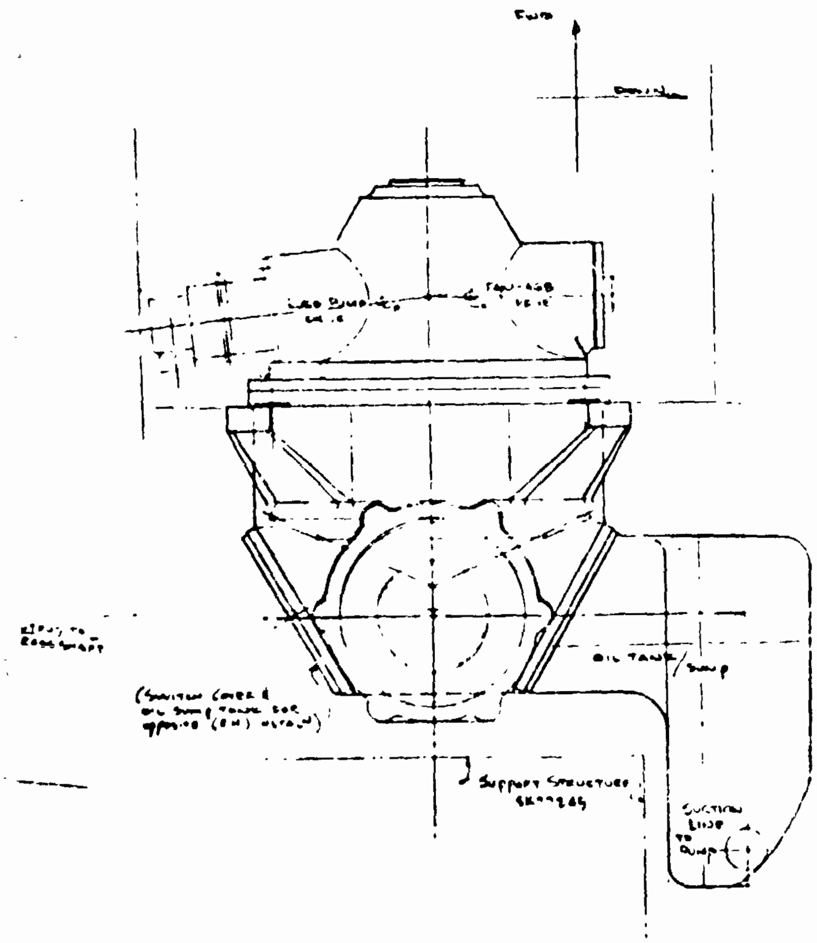

FIGURE 2.13. INTERMEDIATE GEARBOX - EXTERNAL VIEWS OF HOUSING - HTR XV-15

away from the original position to maintain the direction of the accessory drive. Neither change requires unique parts.

A third change depends upon further detail studies. This change would require substitution of opposite-handed bevel gears to maintain identical gear reaction forces in left and right buildups. At some expense in weight, the bearings can be designed to accommodate both loading conditions. There is, therefore, a trade between spares stockage costs and weight between the two approaches. Non-recurring costs would be essentialyy identical for both. As currently designed, it has been assumed that the bevel gears would be replaced when transferring between nacelles.

Gear, bearing and housing materials and fabrication methods duplicate the engine box.

Rotor box (SK-27249), Figures 2.14 and 2.15, gearing consists of a simple, single stage planetary of 5.185 reduction ratio. The four planets are supported on spherical roller bearings, which in turn are located on an extension of the rotor shaft. The internal ring gear is attached to the gear case and upper cover through multiple bolts. The torque reaction is through the bolts to the upper cover, and thence to six mounting lugs to nacelle structure. Forward of the planetary, the rotor loads are transferred to the upper cover through a pair of high-angle, closely spaced tapered roller bearings. This arrangement is a scaled-down version of the planetary and

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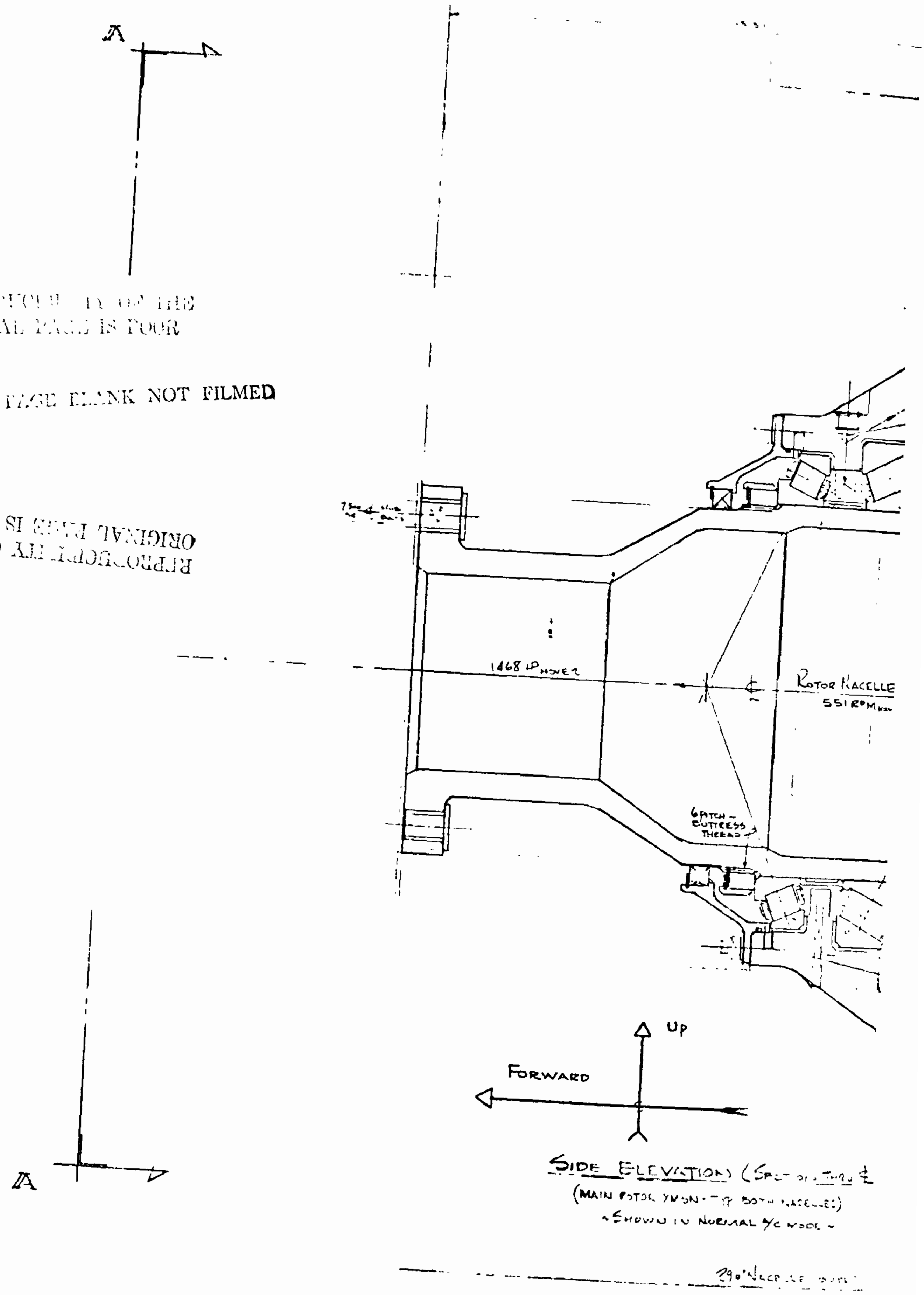

FIGURE 2.14. MAIN ROTC
HTR XV-15

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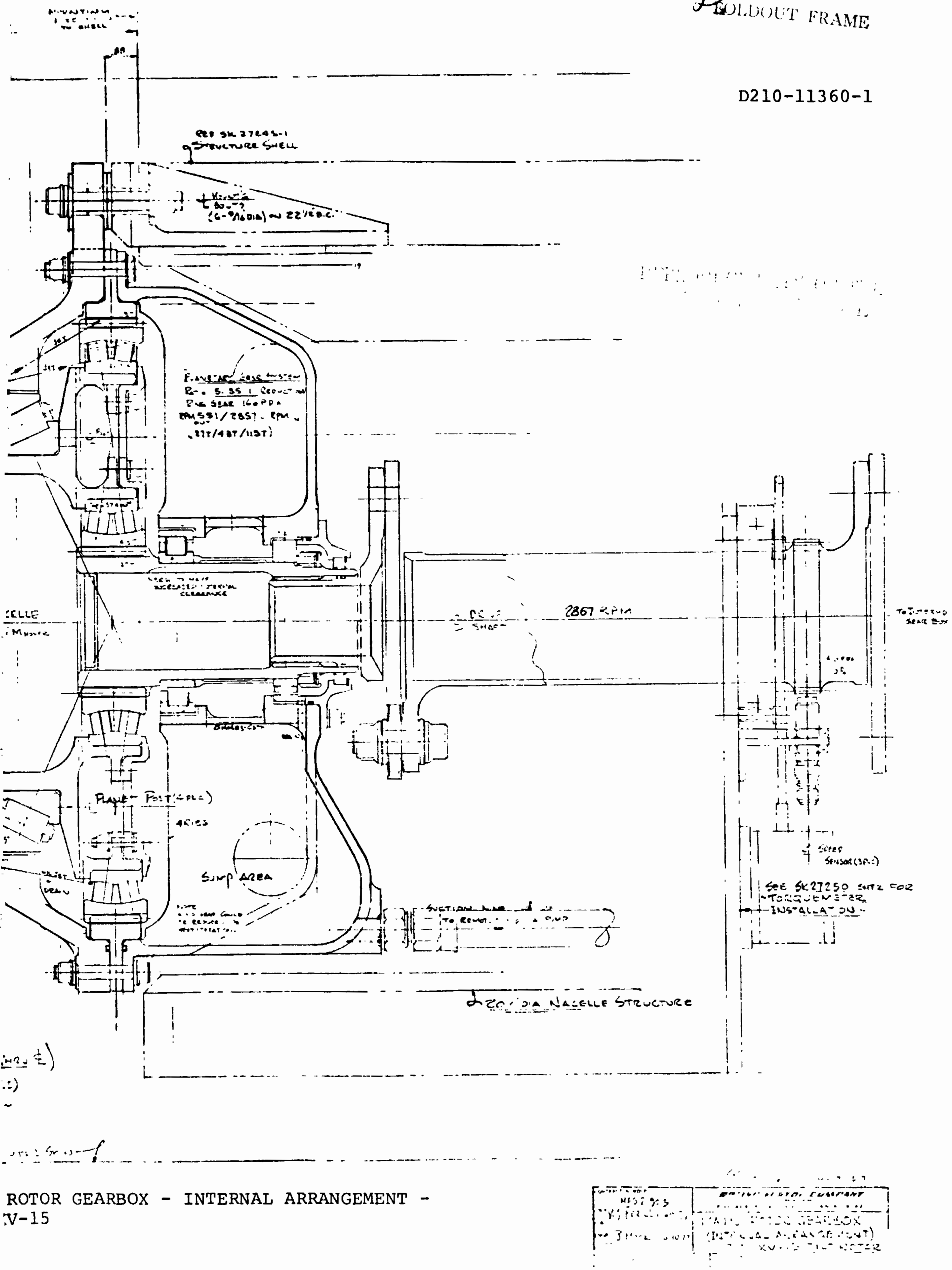

ROTOR GEARBOX - INTERNAL ARRANGEMENT -
V-15

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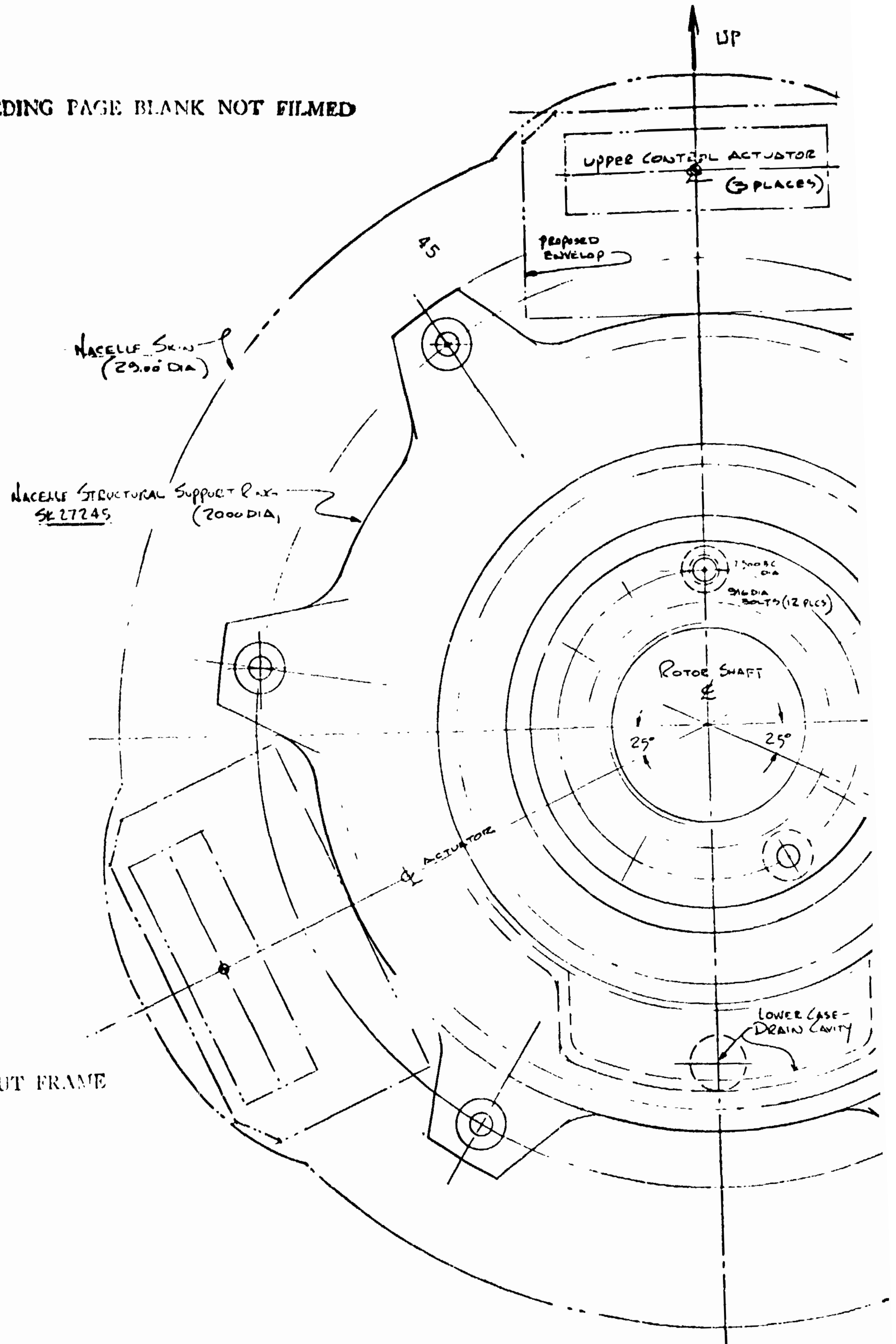

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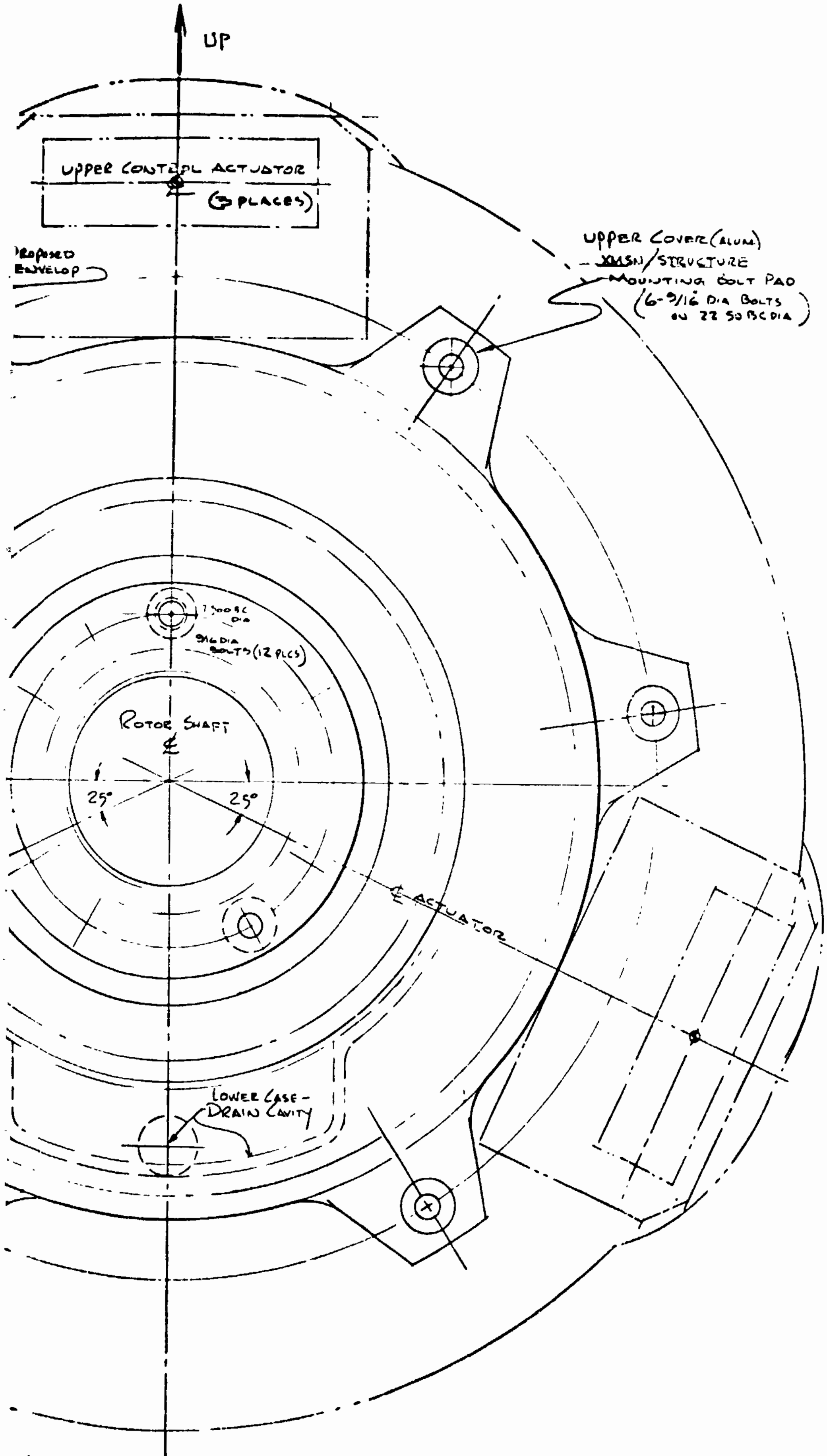
A-A View Looking Aft
(Typical Nacelle Mount)
(Full Size)



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VIEW LOOKING AFT
(TYPICAL ... MOUNT)
(FULL SIZE)

FIGURE 2.15. MAIN ROTOR GEARBOX - MOUNTING ARRANGEMENT - HTR XV-15

rotor shaft successfully flown in the Boeing YUH-61A. At the hub, the interface is a multiple bolt circular flange, also proven in the YUH-61A.

Transfer between left and right nacelles involves a simple translation of the rotor box with no interchange of details.

The upper cover has been designed as a turned part to be made from a forged aluminum pancake billet. To achieve this end, ribs have been eliminated at some expense in structural efficiency, and external hoses connect to lubricators set into the cover. Detail studies to be made will define the weight and cost trades of this approach as compared to a sculptured profile with integral rigs and lube passages. It is believed that considerable cost savings are afforded by the design as shown.

Gear, cast back-housing, and bearing materials and processes are similar to the engine box already described.

Accessory box (SK-27251), Figures 2.16 and 2.17, receives power from a vertical shaft driven by the intermediate transmission. A 90° angle bevel set and a train of spur gears transfer this power to two accessory pads. Pads are designed for 8500 rpm and drive hydraulic pumps and a flight controls generator. An oil cooler blower is driven from the forward end of the bevel shaft and a five-element lubrication pump is driven from the aft end. Struts on the forward face of the accessory box provide the fan and cooler assembly aft support

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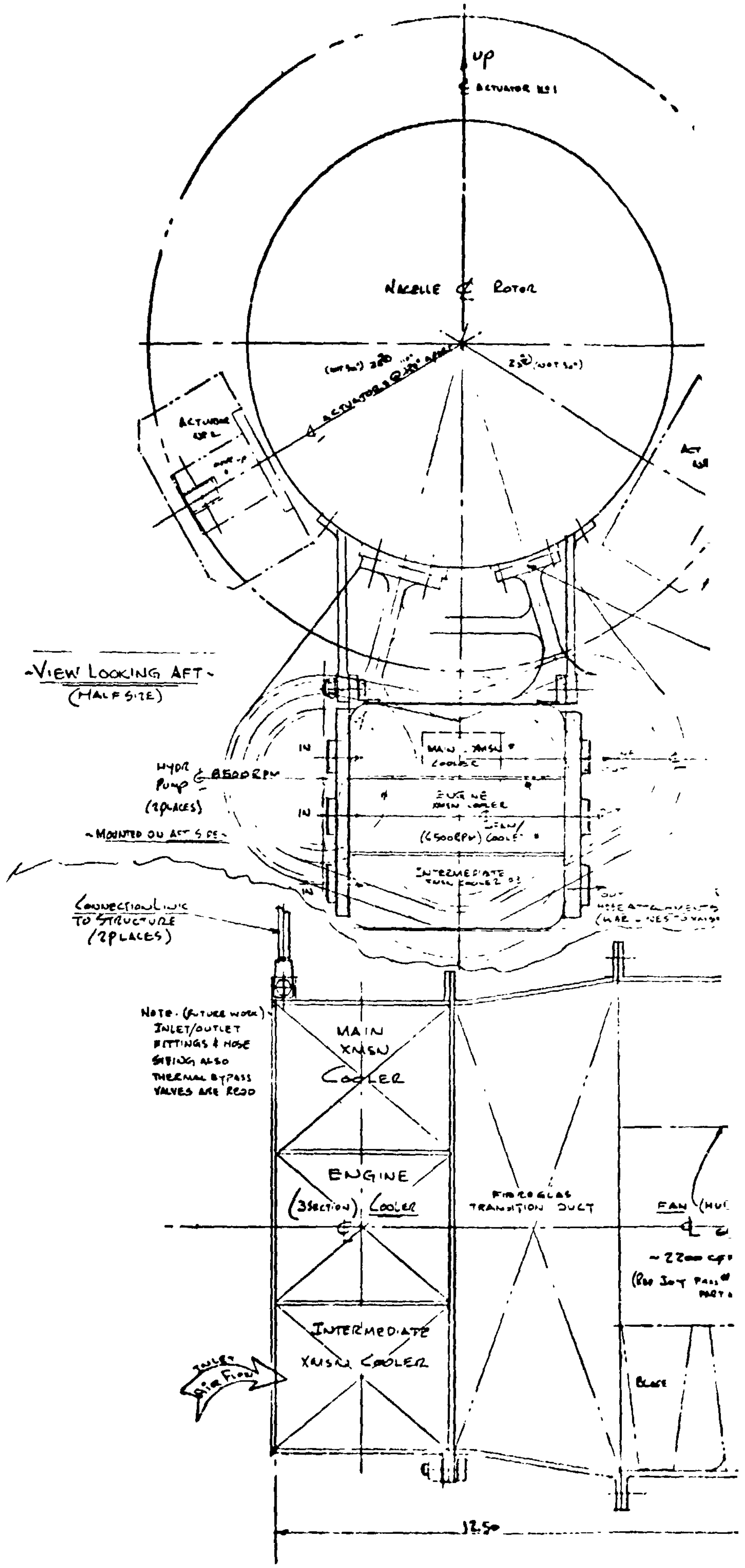

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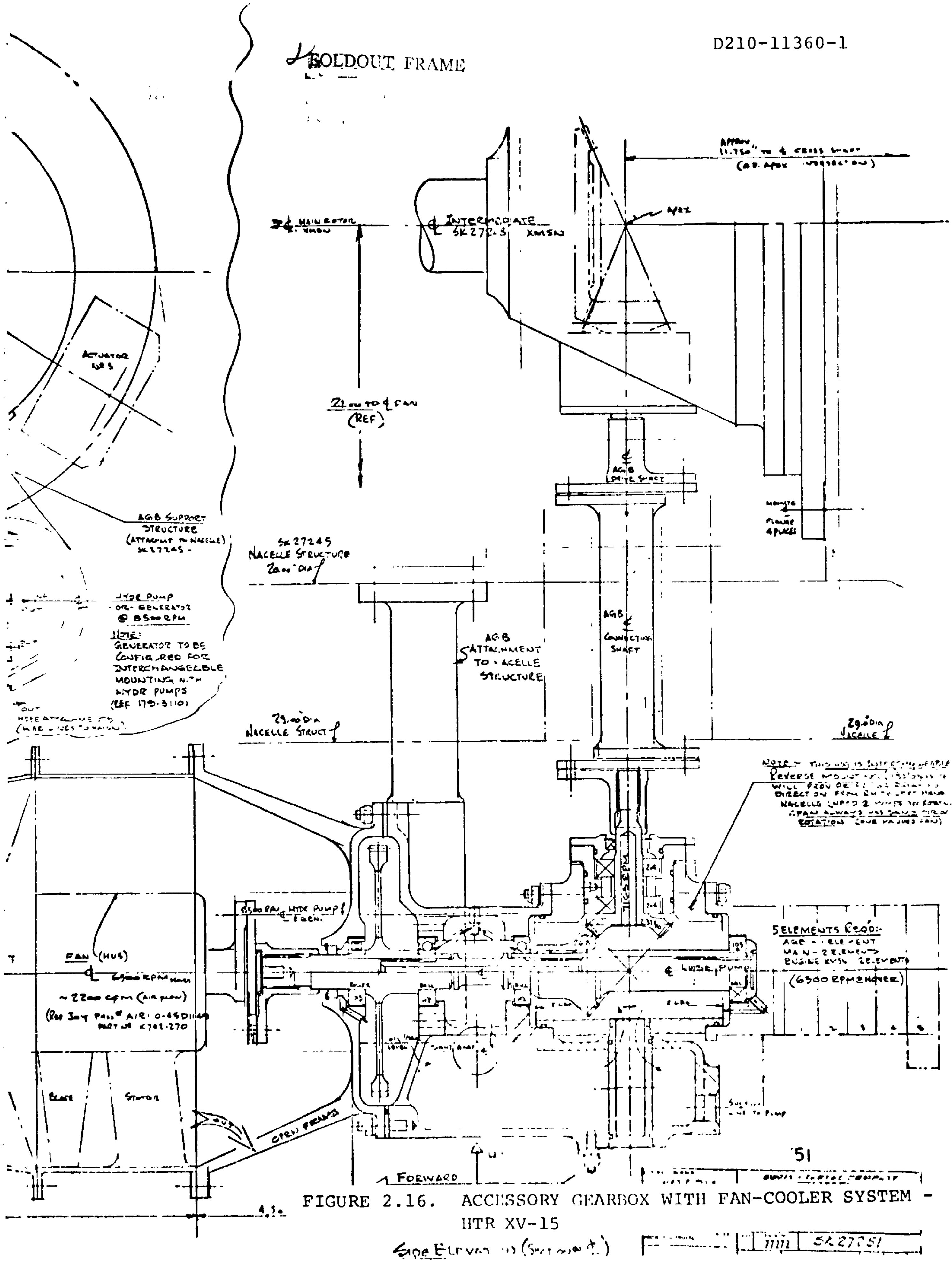
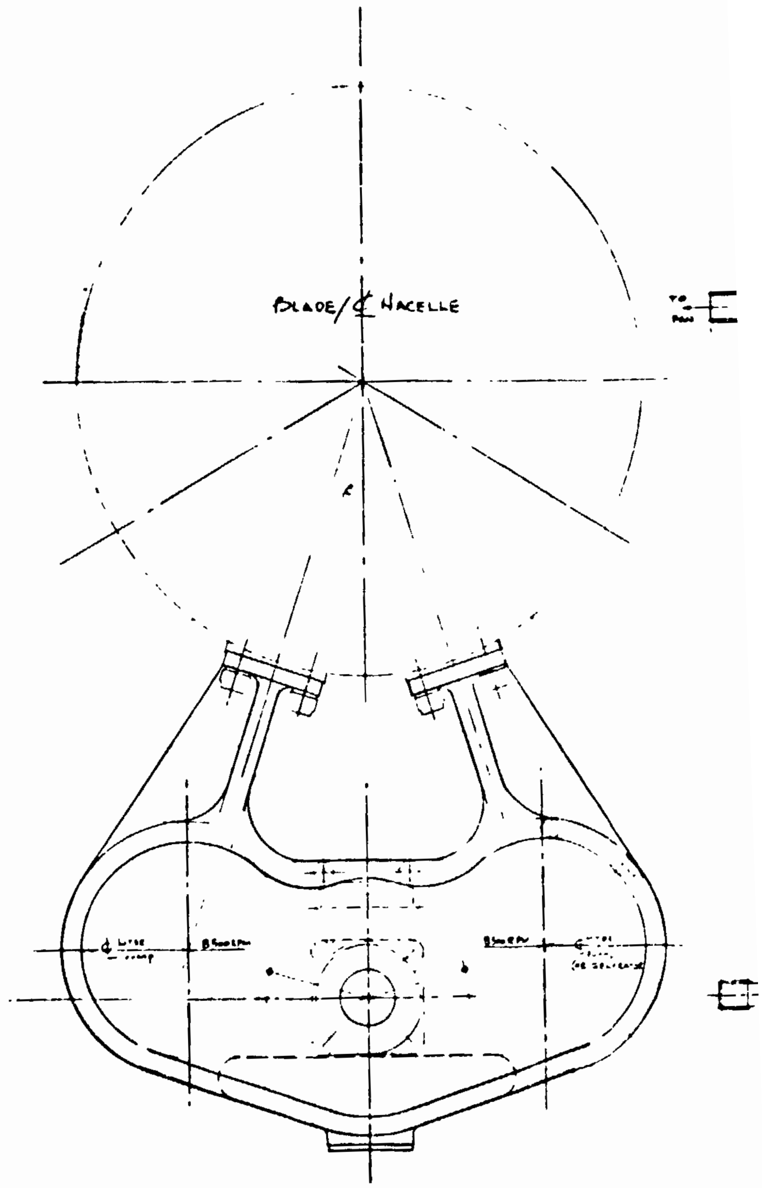


FIGURE 2.16. ACCESSORY GEARBOX WITH FAN-COOLER SYSTEM - HTR XV-15

Side Elevation (Shot on ¢)



- LOOKING AFT -

FIGURE 2.17

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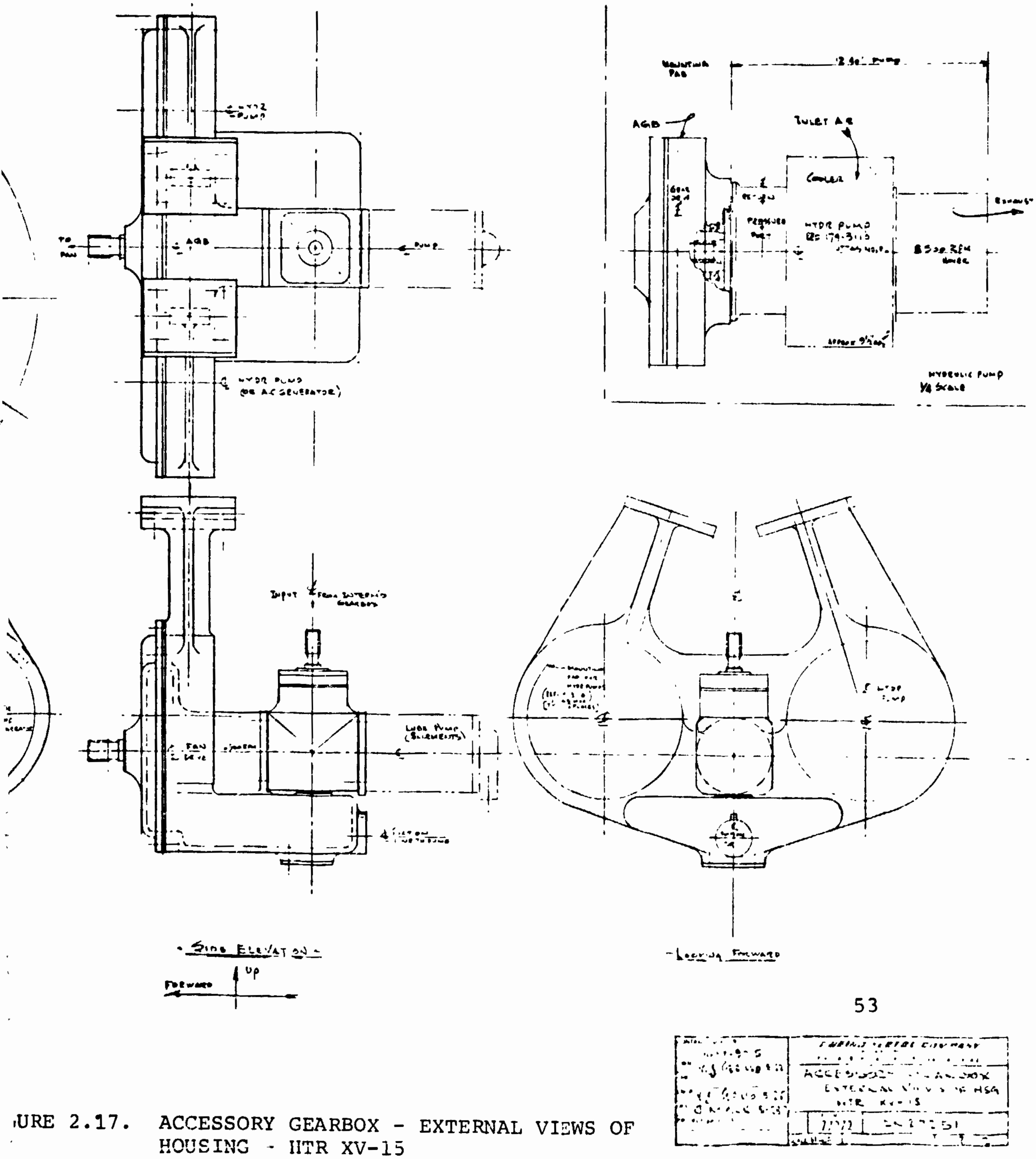

FIGURE 2.17. ACCESSORY GEARBOX - EXTERNAL VIEWS OF HOUSING - HTR XV-15

and axial locator. Vertical struts support the box itself from the nacelle structure. The accessory box has been designed for maximum accessability and ease of maintenance of the driven accessories.

Transfer from left to right nacelle is accomplished by disconnecting and reassembling the bevel gear portion, at the same time reversing the lube pump mount from one end to the other. This assembly task, which requires no internal adjustments, is necessary to preserve identical pad rotations in either nacelle.

Component materials and processes are similar to that of the engine and other boxes.

2.3.2.3 Shafting, Couplings and Torquemeters

The new portion of the drive shafting connects engine box to intermediate, intermediate to rotor and intermediate to an interface with existing cross-shafting inboard of the nacelle (SK-27250), Figures 2.18 and 2.19. A short section of new shaft connects intermediate box to accessory drive gearbox.

New shafting follows conventional Boeing Vertol helicopter design practice in that the shaft is a thin wall, 2024 aluminum, tube, rivetted to end adapters which in turn connect to multi-plate metal flexible couplings of the Thomas type. This coupling affords excellent redundancy features, low cost and weight. It is limited in misalignment

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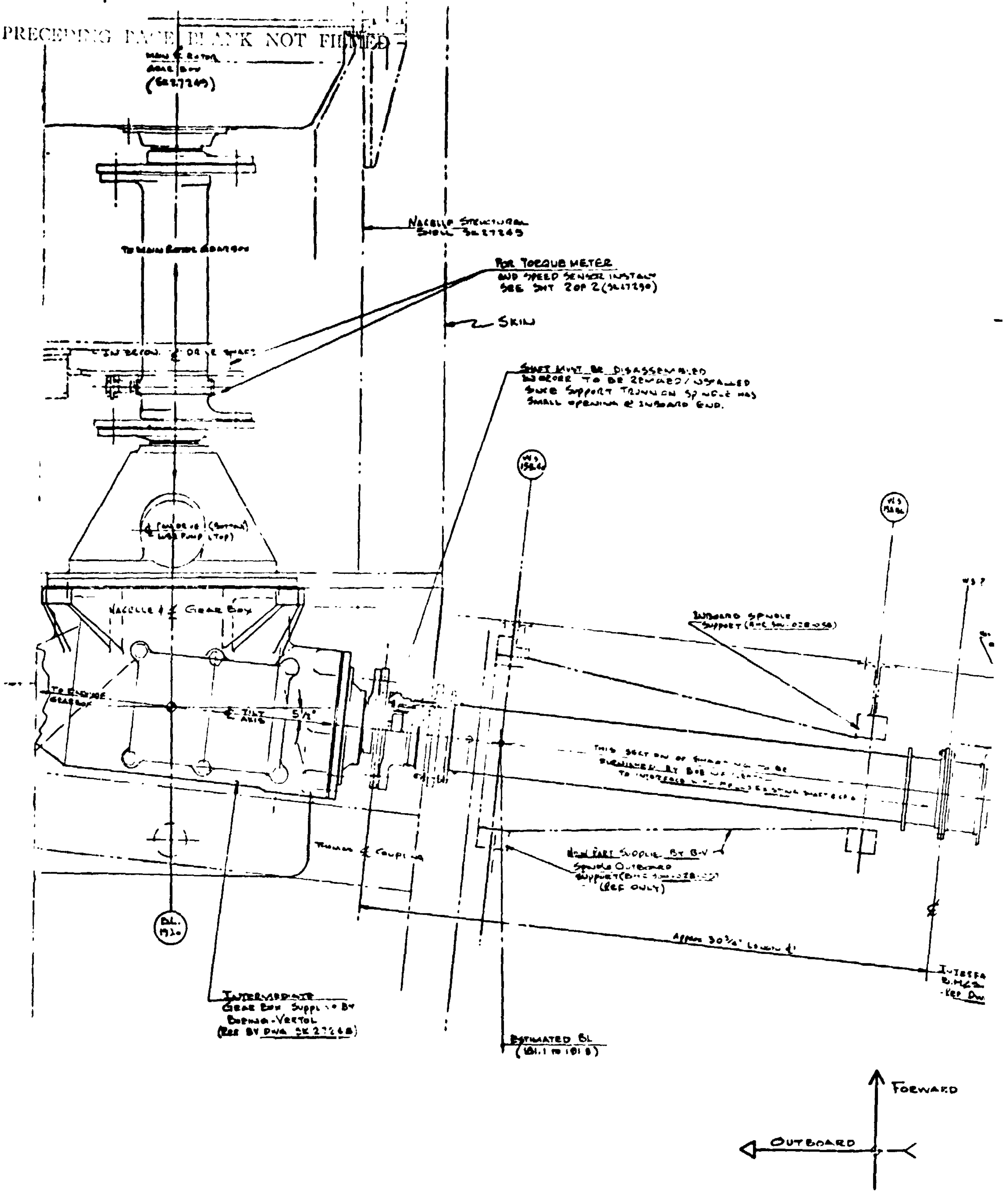

PLAN VIEW
CROSS SHAFT INSTALLATION

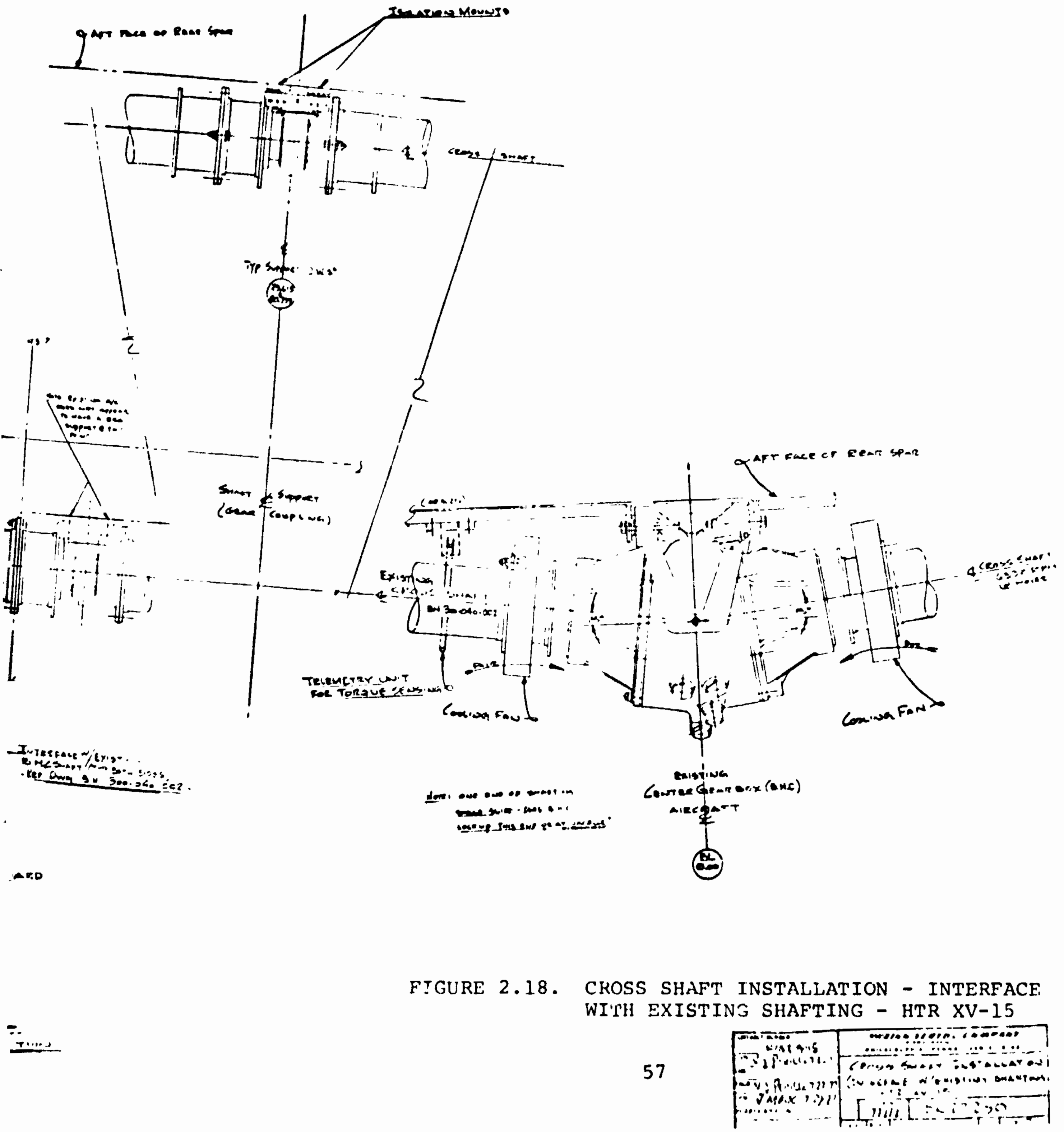

FIGURE 2.18. CROSS SHAFT INSTALLATION - INTERFACE WITH EXISTING SHAFTING - HTR XV-15

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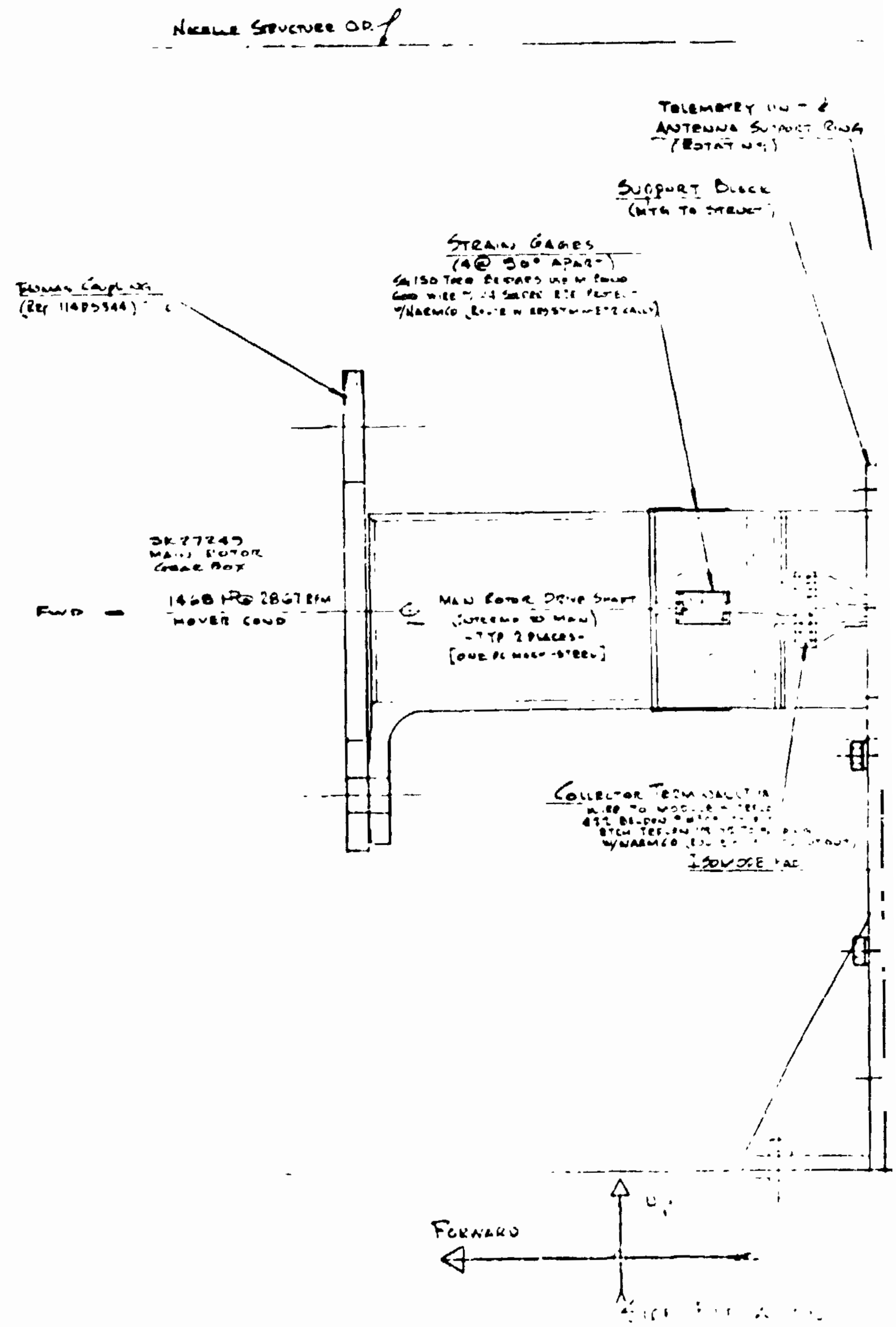

FIGURE 2.19. CROSS
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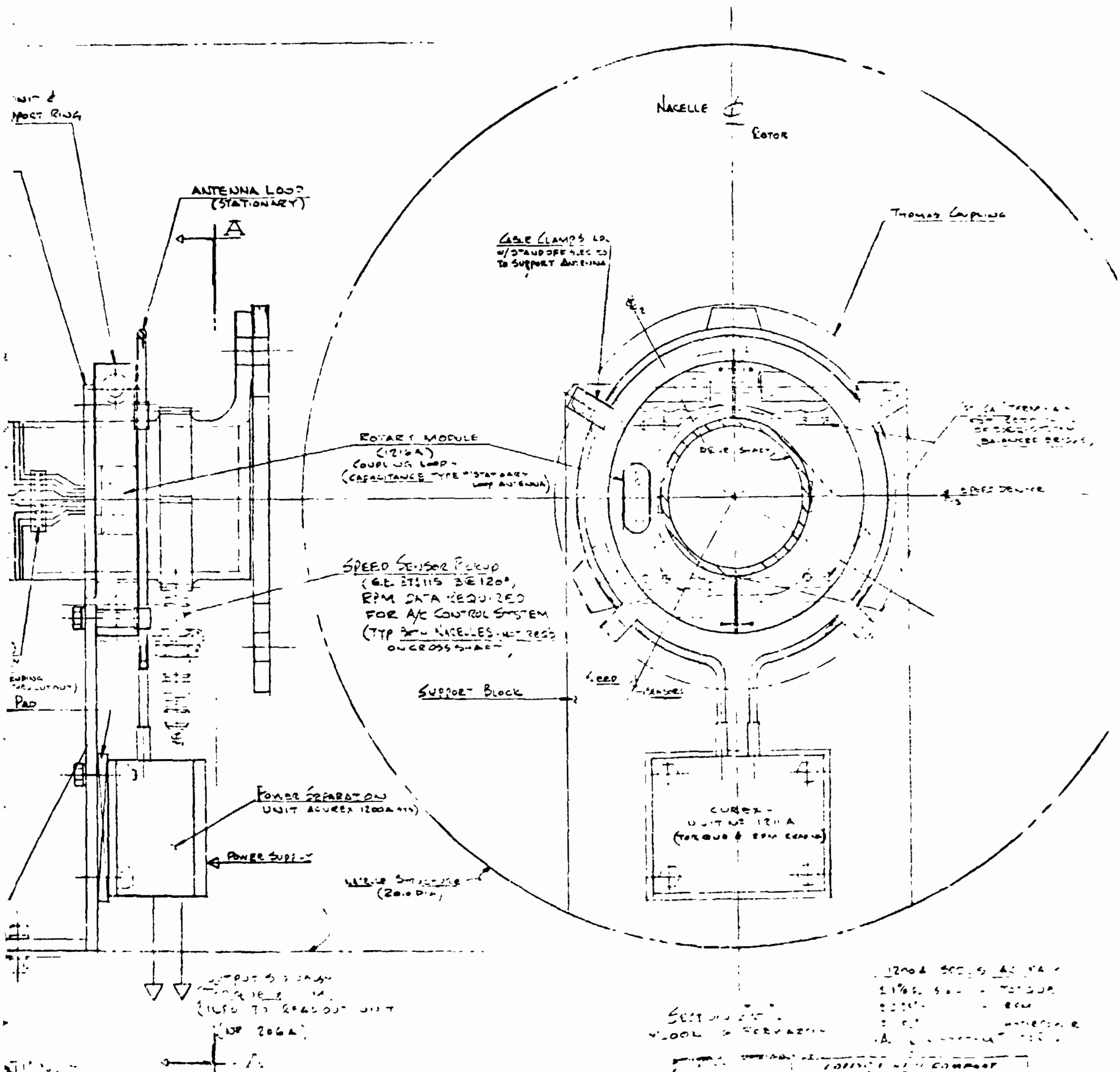

CROSS SHAFT INSTALLATION - AIRCRAFT
TELEMETRY INSTALLATION FOR MAIN SHAFT
AND CROSS SHAFT - HTR XV-15

59

capacity to the order of one degree. Further study with a structural deflection analysis will be required to substantiate the use of this coupling in all locations. An accepted alternative is the metal diaphragm (Bendix type) which, by multiple diaphragms, can practically accommodate misalighments to 5 degrees.

No intermediate bearing supports are required between gearboxes in the nacelle since the spans are sufficiently short to maintain subcritical shaft operation. At the interface with existing wing cross-shafting, insufficient design definition is available at this time to completely identify the bearing and coupling requirement. It is probable, however, that a shaft support bearing will be required at this point to assure subcritical operation and misalignment capability. Passage of a new shaft section through the nacelle pivot housing bore represents a design constraint only to the extent that the inboard coupling must be detachable from the shaft to make the necessary clearance diameter.

Coupling designs and sizes are taken throughout from existing Boeing Vertol designs that meet or exceed the required torque capability.

Engine box to intermediate shaft section are unique to each nacelle; intermediate to rotor and to accessory boxes are identical left to right. As noted, the exact length and interface design of the intermediate to cross-shaft section has not been finally determined.

There are three torquemeters in an aircraft assembly; one on each intermediate to rotor box shaft section and one on the cross shaft. All torquemeters are strain gage type, powered by induced current and readout by telemetered radio frequencies through a common loop antenna surrounding the shaft. Advantages of this type of torque sensing are ease of incorporation, light weight, high accuracy. Accuracy and reliability have been demonstrated in previous Boeing Vertol installations in test equipment, hydrofoil power trains and inside transmissions. The equipment has been supplied by Accurex, Inc., to Boeing specifications for these applications.

2.3.2.4 Lubrication, Cooling and Condition Monitoring

Lubrication systems are separated for each gearbox, thus providing ready identification of the location of any abnormality evidenced by pressure, temperature or debris monitors and also eliminating the possibility of cross-contamination between boxes. The design concept of the main power boxes is similar in that the oil, having passed through the box, is drawn through a full-flow metallic debris sensor and is them pumped through a filter and a cooler to return to the gearbox through a jet protection screen (see Schematic Diagram SK-27258, Figures 2.20, 2.21 and 2.22). The rotor and engine boxes are mechanically scavenged and the oil fed to

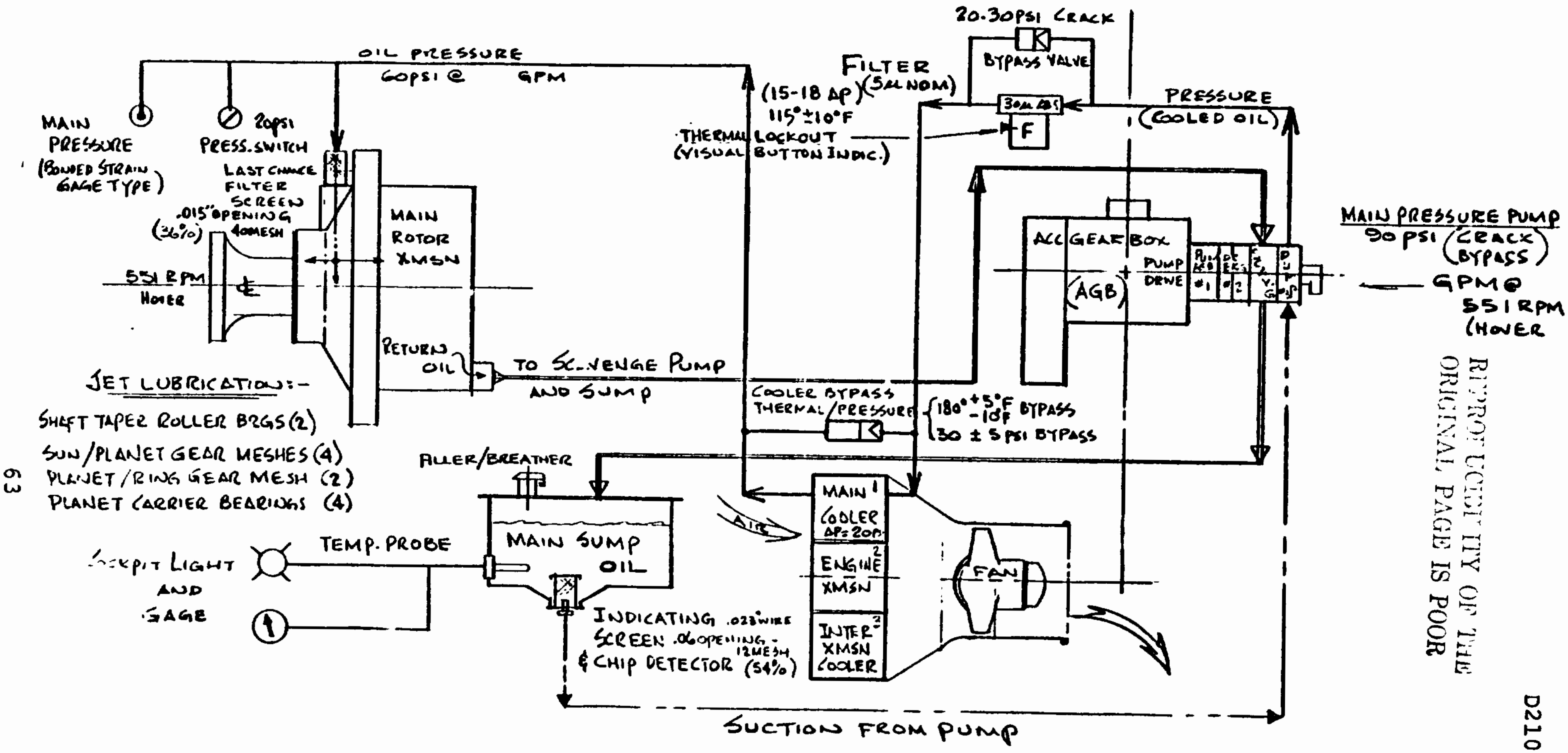

FIGURE 2.20. LUBRICATION SCHEMATIC (HTR XV-15) - MAIN ROTOR TRANSMISSION SYSTEM - SK 27258, SHEET 1 OF 3

SK 27258
SHT 1 OF 3

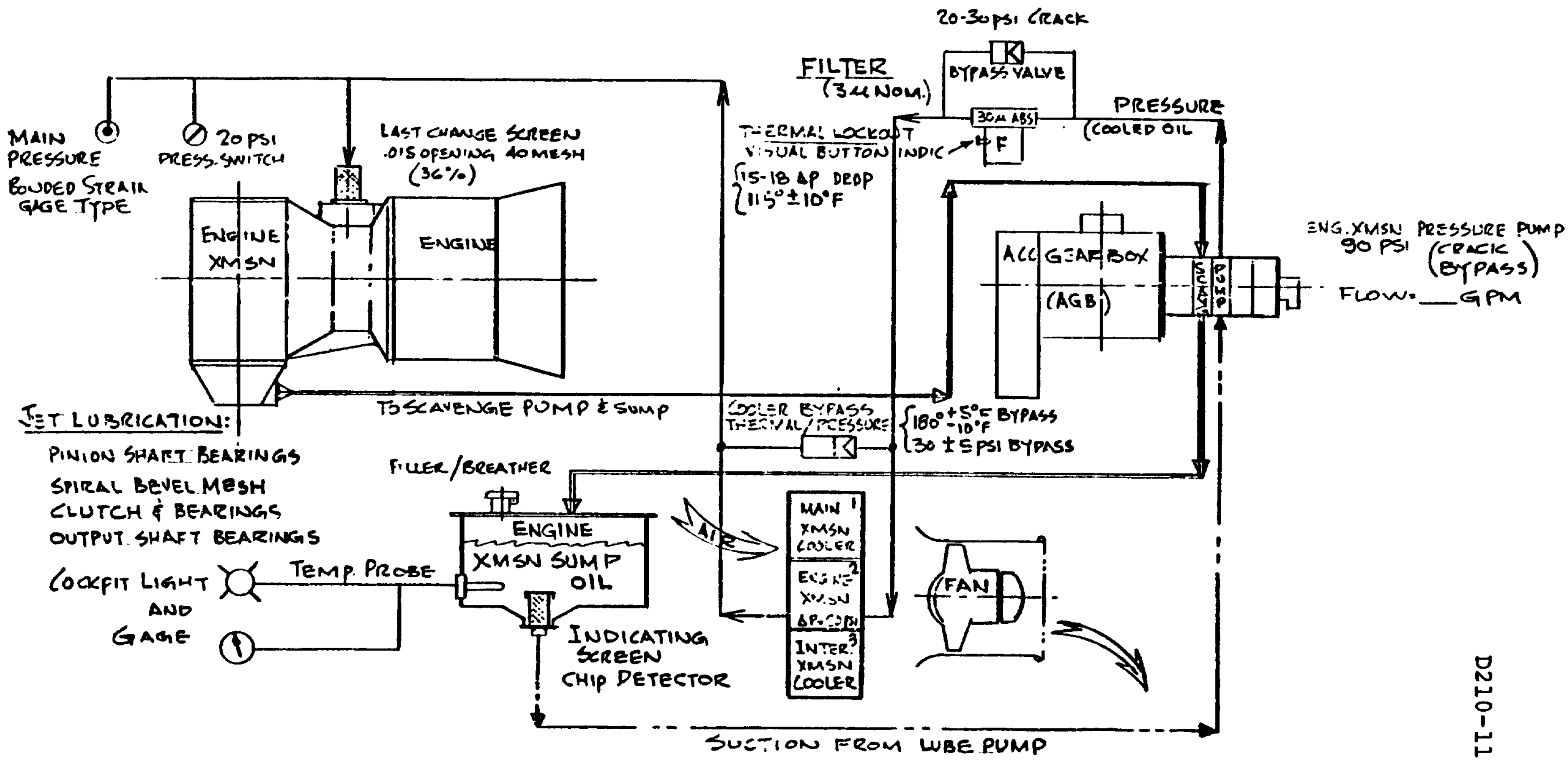

FIGURE 2.21. LUBRICATION SCHEMATIC (HTR XV-15) - ENGINE TRANSMISSION SYSTEM - SK 27258, SHEET 2 OF 3

SK 27258
SHT 2 OF 3

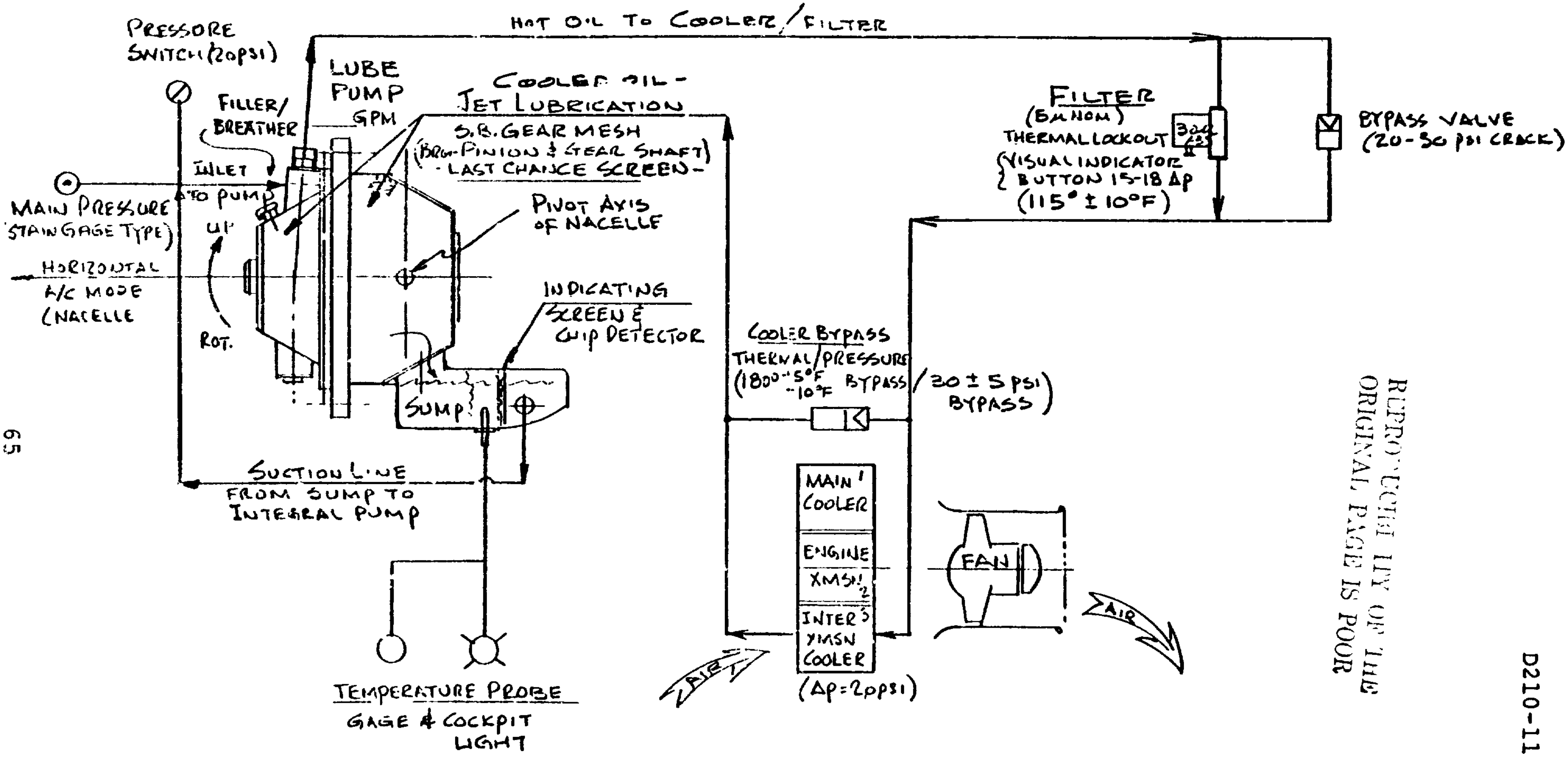

FIGURE 2.22. LUBRICATION SCHEMATIC (HTR XV-15) - INTERMEDIATE TRANSMISSION SYSTEM - SK 27258, SHEET 3 OF 3

SK 27258 SHT 3 OF 3

airframe-mounted tanks. The intermediate box drains by gravity to a pistol-grip shaped sump below and behind the gears. This provides drainage in either nacelle attitude and at intermediate points as well. Space constraints precluded gravity drainage in the other two boxes.

The accessory box lubrication system is simplified because no cooler is needed to dissipate the small amount of heat generated by the low (60-80 hp) transmitted power. Oil is gravity drained to a sump and is them pumped through a filter and thence to the lubrication jets.

Five lube pumps in a common housing are driven from the aft end of the accessory box. Two each scavenge and feed rotor and engine boxes and the remaining pump services the accessory box itself. The intermediate box mounts and drives a single element feed pump. All pumps are conventional Gerotor positive-displacement type with relief valves. Pump capacity is sized to provide required flow ar cruise rpm, with relief valve regulation at hover rpm.

Filters are full flow, 30 micron, disposable media type with a clogged-filter bypass valve. A delta-pressure indicator on each filter provides warning of an impending bypass, and thus allos corrective action before downstream contamination occurs.

Full-flow debris detection in the form of an indicating screen with conductive wires provides warning of metallic particles generated in the transmission.

Air-oil coolers are mounted in a common frame to service the engine, rotor and intermediate transmissions. Analysis of the cooling requirements at hover and cruise fan rpm's and consequent air-flows has been applied to the sizing of cooler and fan. Each cooler includes a high pressure by-pass to prevent over-pressures in the core in cold weather starts.

A single fan in each nacelle induces air flow through the coolers. The fan is of standard aircraft design. It is self supported on greased ball bearings, and is mounted through stators and struts to the forward face of the driving gearbox. It is likely that in cruise flight ram induction would be sufficient to make the fan superfluous, and it may in fact be desirable to provide pressure relief doors in the fan duct to prevent overpressure in this area.

Oil pressure is monitored on the gearbox side of the jet protection screen so that any clogging at this non-bypassable screen is detectable. The function of the jet protection screen is to prevent any material lodged in external lines or cooler, or bypassed around the filter, from clogging lube jets. The screen opening is sized accordingly. As a further safeguard against local loss of lubrication, jets are multiple to the same bearing.

Oil temperature is monitored by pickups located in sump or tanks, measuring hot oil out of the box and before cooling.

Oil levels are monitored at servicing intervals by sight glasses in each sump or tank. During the same servicing, the filter bypass indication will be inspected.

The general and detail design concepts, materials and processes used in the lubrication, cooling and condition monitoring systems ahve been developed and demonstrated on Boeing Vertol helicopters. In common with the gearbox and shaft design, they represent a low-risk approach to the problem. Early attainment of high reliability levels can, therefore, be predicted.

2.3.3 Flight Control System

The selected fly-by-wire control system configuration for XV-15 consists of a triplex self-monitored analog Primary Flight Control System (PFCS) interfaced with a dual analog Stability and Control Augmentation System (SCAS). The SCAS provides for stability and gust alleviation inputs to the PFCS.

Digital computation was considered for both the PFCS and the SCAS; an analog mechanization was selected for PFCS to minimize program risks related to electromagnetic interference tolerance, configuration control, development cost and several other factors. The complexity of the SCAS (by itself) did not warrant a digital mechanization.

This study drew on a previous system design study conducted for the UH-61A helicopter (Boeing production Utility Tactical Transport Aircraft System (UTTAS) entry). This design represented a refinement of the Heavy Lift Helicopter (HLH) system design. Refinements centered on the actuator design. By using magnetic summing in the actuator electrohydraulic valve instead of force summing, the actuator was simplified to eliminate one control stage piston and the need for one full time hydraulic system. Details on the selected actuator configuration are given in Volume II, Appendix I.

System transfer functions were based on previous studies as defined in Reference 1. Physical characteristics and limitations of the equipment were defined by configuration requirements in this study (i.e., definition of new nacelle and the need to interface with existing XV-15 systems. Consideration of the XV-15 interfaces, in particular cockpit controls and airplane surface actuator installation, was limited by availability of Bell drawings. Table 2.3 gives a summary of the specifications established for the system. The full specifications for the system are defined in Appendix III.

2.3.3.1 Overall System Description

Studies leading to the original HLH system design considered two system configurations as shown in Figure 2.23. The two concepts differ in the method of interfacing the pilot and automatic control loops to the actuators.

In the first case the pilot and stabilization signals are mixed in a central computer. There is no direct pilot control of the actuators; failure of the stabilization loops result in failure of the pilot control path. The computer would typically be digital. This approach was used for the Tactical Air Guidance System (TAGS) (flight control system) which was a triplex digital system flown with a mechanical backup on the CH-47.

The second approach is more directly analogous to the systems found in current aircraft. The primary control system provides for direct pilot control of the actuators via relatively simple path. Assuming the vehicle can be flown without augmentation, flight safety is vested in this path. Augmentation is via a

OVERALL SYSTEM ACCURACY	2%
SYSTEM NULL	.020 INCH ACTUATOR
RESOLUTION	.002 INCH ACTUATOR
HYSTERISIS	.004 INCH ACTUATOR
ROTOR ACTUATOR HYDRAULICS	40 RAD/SEC, .7 DAMPING FACTOR
FLIGHT SAFETY RELIABILITY	.9999999 FOR TWO-HOUR MISSION
ROTOR CONTROL ACTUATOR	±3.00 INCH STROKE
	2.50 IN./SEC NO LOAD VELOCITY
	±5,200 LBS STALL
DESIGN SPECIFICATIONS	MIL-E-5400 ELECTRONIC EQUIPMENT
	MIL-C-5503, MIL-H-5440, MIL-H-8775 HYDROMECHANICAL EQUIPMENT

E 2. ?-15 FLIGHT CONTROL SYSTEM - SPECIFICATION SUMMARY

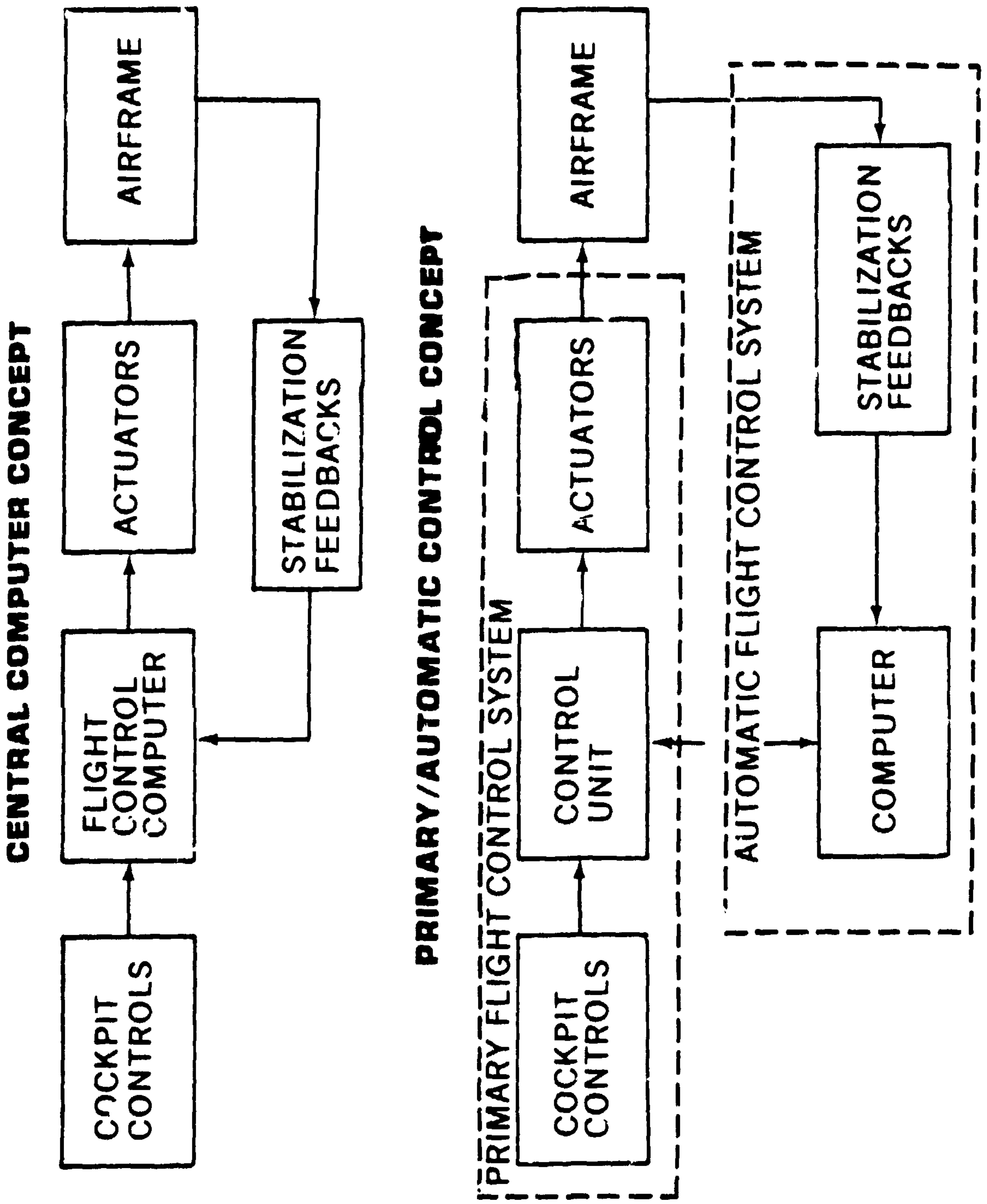

FIGURE 2.23. OVERALL SYSTEM CONFIGURATION

separate path with redundancy suited to mission requirements. Failures of this path do not cause failures of the primary system. This approach was selected for HLH and UTTAS and is recommended for XV-15.

The fly-by-wire primary flight control system controls the rotors, flaperons, rudder, and the elevator. It is comprised of conventional dual pilot station cockpit controls, a triple redundant electrical link between cockpit controls and actuators, and dual redundant actuators. The system is powered by a dual hydraulic supply and a triple electrical supply which is derived from the normal dual electrical supply and an additional transmission mounted direct current generator. Figure 2.24 illustrates the basic components of the system.

Cockpit Controls

Cockpit controls at each pilot station consist of a cyclic stick providing longitudinal and lateral control, pedals for directional control, and a throttle control lever. The conventional cockpit control configuration is chosen rather than the side arm controller configuration to provide better pilot ingress/egress, easier synchronization between pilot and copilot, and reduced pilot workload without stability and control augmentation.

Electrical Link

The electrical link is triplex to meet safety and mission reliability goals. Each channel of the direct electrical

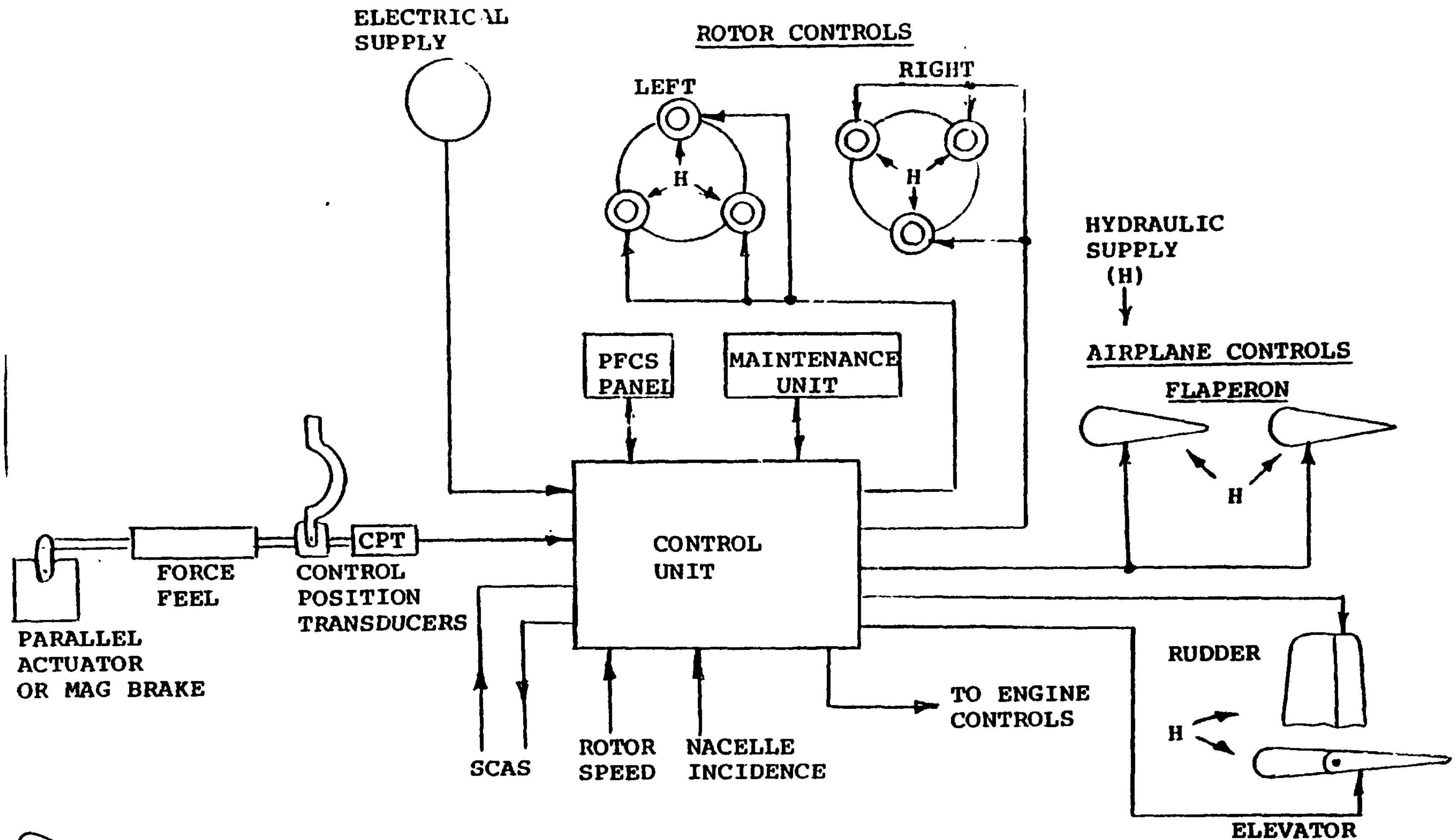

FIGURE 2.24. FLY-BY-WIRE PRIMARY FLIGHT CONTROL SYSTEM

C-2

link includes Control Position Transducers (CPT), a Control Unit, a Junction Box, and associated electrical cables.

The Control Position Transducers translate cockpit control motions into electrical analog signals which, in turn, are transmitted to the Control Unit. The Control Unit performs the equivalent functions of the mechanical flight control linkage; specifically, control signal mixing, limiting and gain scheduling with nacelle position. The Control Unit also provides SCAS signal integration into the primary flight control system, redundancy management, and fault isolation, control of engine performance.

The selected redundancy management scheme encompasses independent in-line self-monitoring of each channel. This concept permits the triplex system to be dual fail functional because each channel detects its own failures independently from the others. Independence of control channels permits considerable simplification of the control logic and prevents any possible failure propagation or electrical interference between channels.

The cables connecting the system elements are preformed and shielded multi-conductor type, providing point-to-point connection at rugged self-locking threaded connectors with strain relief. Each cable is a line replaceable unit.

Provision for adjusting rotor rpm, rotor torque balance, resetting a channel in flight and clearing stored channel failure information is made at the Primary Flight Control System Panel.

The Maintenance Unit provides automated checkout of the entire system in approximately two minutes; if a failure is detected, it provides the necessary failure information to quickly locate the failed line-replaceable unit.

Actuators

The electrical link controls functionally identical electro-hydraulic actuators at each location. The actuators have dual output rams controlled by a conventional tandem hydro-mechanical servovalve (called the power stage). The servo-valve is driven by a dual electrohydraulic actuator (called the control stage) which interfaces with the electronics. The control stage of both the rotor and airplane surface actuator is idnetical. The area of the power stage valve slots and output pistons is configured to suit the rotor or airplane application. (Two sizes will be built).

2.3.3.2 System Transfer Functions/Mechanization

Required system transfer functions and performance characteristics are defined in specification, Appendix III. Block diagrams from the system specification along with details on the selected hardware mechanization of the specified transfer functions as proposed by General Electric Aerospace Control Systems Department is given in Volume II, Appendix I.

2.3.3.3 Redundancy Management

Since a single channel will not provide the reliability/survivability characteristics necessary to meet the primary flight control system objectives, it is necessary to provide redundant channels.

The Primary Flight Control System is a triple redundant self-monitored electrical link controlling dual redundant electrohydraulic actuators.

As shown in Figure 2.25, the self-monitored concept of redundancy management for each channel involves use of two identical signal paths in each channel between the cockpit controls and the actuator input. If a discrepancy occurs which is greater than a pre-established tolerance level, that channel is considered to have failed and is shut down. The electrohydraulic actuators have dual hydraulic sections and triplex electrical sections. Thus, the mechanical linkage of the present control system is replaced with a triplex electrical link, while the present dual hydraulic power section is maintained in the fly-by-wire mechanization. Each channel of the electrical link is powered by an independent electrical supply.

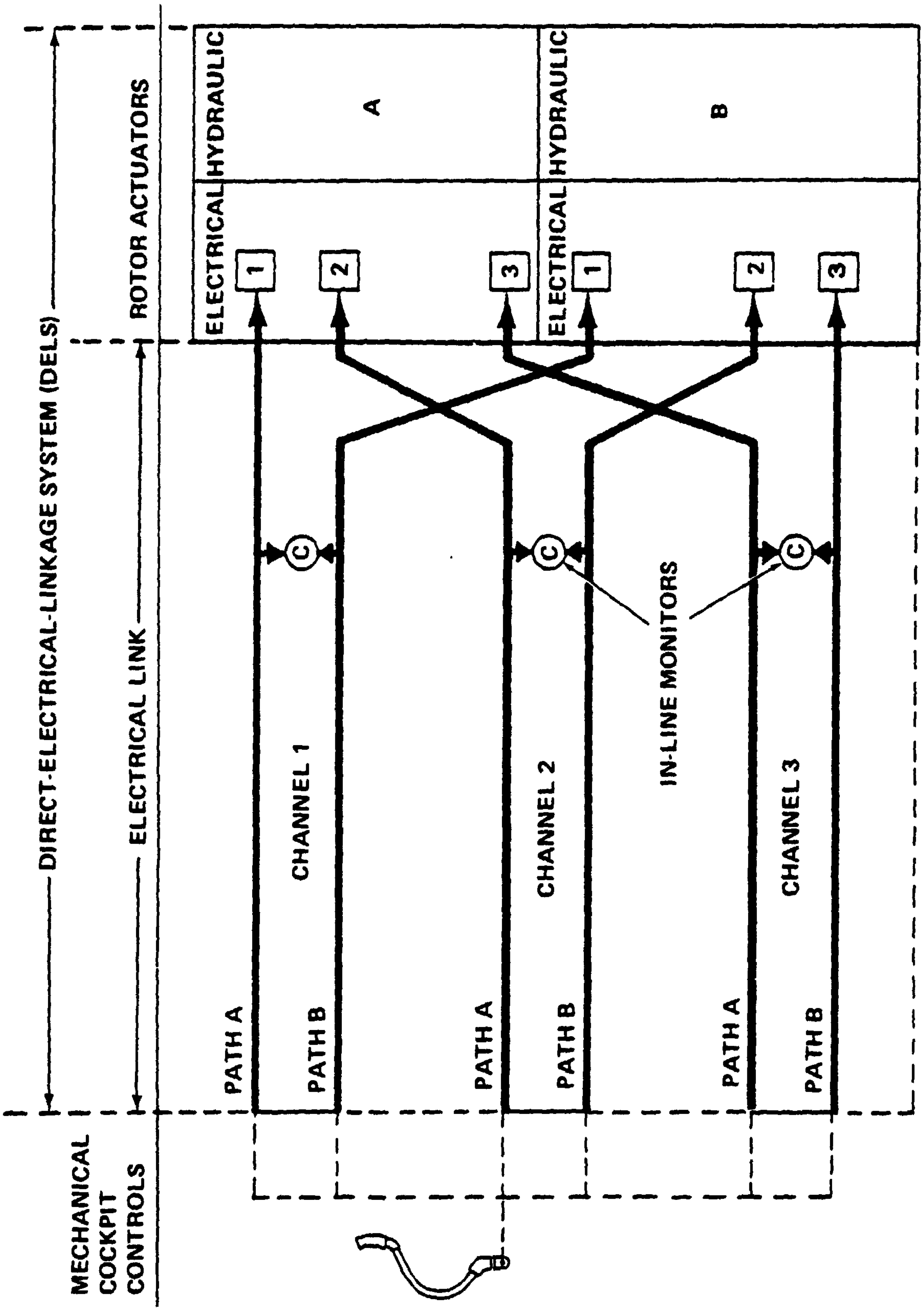

FIGURE 2.25. PRIMARY FLIGHT CONTROL SYSTEM - REDUNDANCY MANAGEMENT

Tracking of three channels is maintained by control of overall gain tolerances. Channel inputs to the actuator are summed magnetically in the electrohydraulic valve of the actuator. The resulting actuator position is the average of all six signal path signals (2 each in 3 channels). Analysis and breadboard test of the magnetic summing did not indicate a need for interchannel compensation; however, an active, on-line scheme such as used on the HLH program could be used to equalize the channels, if necessary. Inherent failsafety without time-critical switching is maintained for first failures by use of magnetic summing. After channel shutdown the output position will be the average of the four remaining paths.

The electrical link inputs are the cockpit controls, the SCAS, and nacelle position. Each channel receives the same signal from the self-monitored cockpit Control Position Transducers; this reduces the number of transducers required per axis from 6 to 3 in comparison to the fully dualized input used on HLH. Figure 2.26 shows a typical one-axis channel of the triplex electrical link up to the actuator servo loop.

SCAS commands are distributed to all three channels. Signals are processed through authority/rate limited interfaces in the Control Unit and sent to both servo loops. Using this approach, SCAS failures cannot cause primary system channel shut downs.

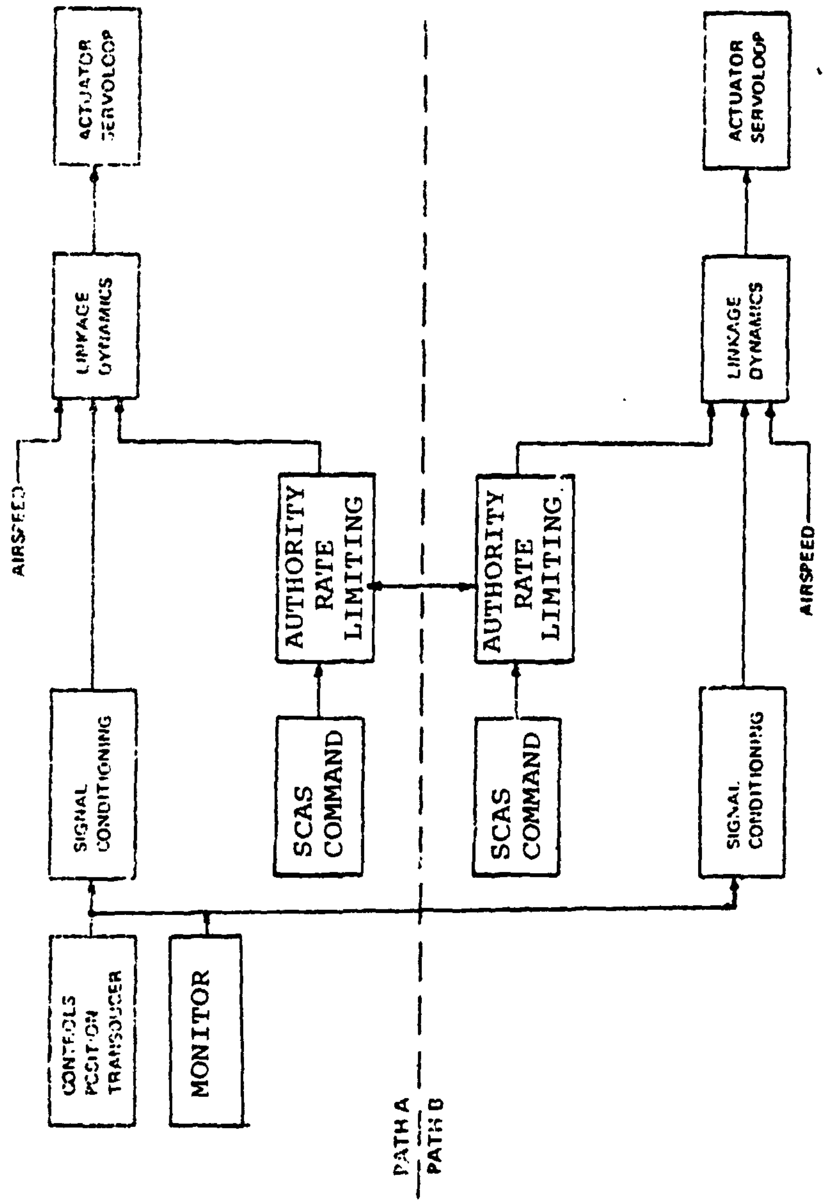

FIGURE 2.26. DUAL PATH PRIMARY FLIGHT CONTROL CHANNEL

Therefore, the primary control system reliability becomes independent of SCAS. Placing the SCAS authority and/or rate limits within the primary system results in aircraft transients after SCAS failures, being less than or equal to those with the mechanical system. Figures 2.27 and 2.28 show the mechanization of the SCAS and GLAS interfaces. Pilot warning without shutdown will be provided if the channel outputs differ. Pilot can then shut down these inputs, if necessary.

The actuator failure detection mechanication is shown in Figure 2.29. Paths A and B summing amplifier outputs are compared to detect failures in the feed forward and feedback electronics. After comparison, these signals are averaged in the servo amplifiers. This removes any accumulated error due to tolerance stack up from the next stage of comparison.

Electrohydraulic valve currents of two actuator sections are compared. This comparison will detect failures in the servo amplifier, actuator wiring, or EHV torque motor coil. When failures are detected, the channel's input to the electrohydraulic valves is removed by opening a relay contact and the pilot is notified.

Failures of the electrohydraulic valve downstream of the torque-motor are detected by comparison of valve current to control stage piston differential pressure. Of particular concern are those failures which would cause an EHV hardover

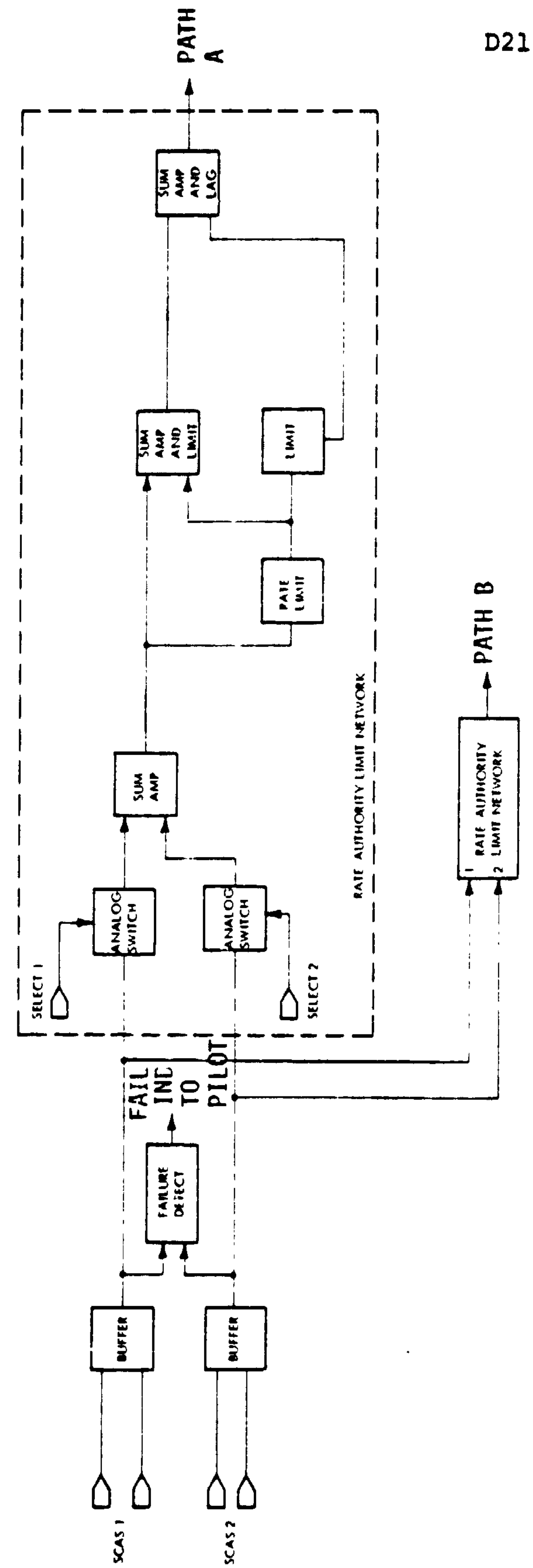

FIGURE 2.27. SCAS INTERFACE

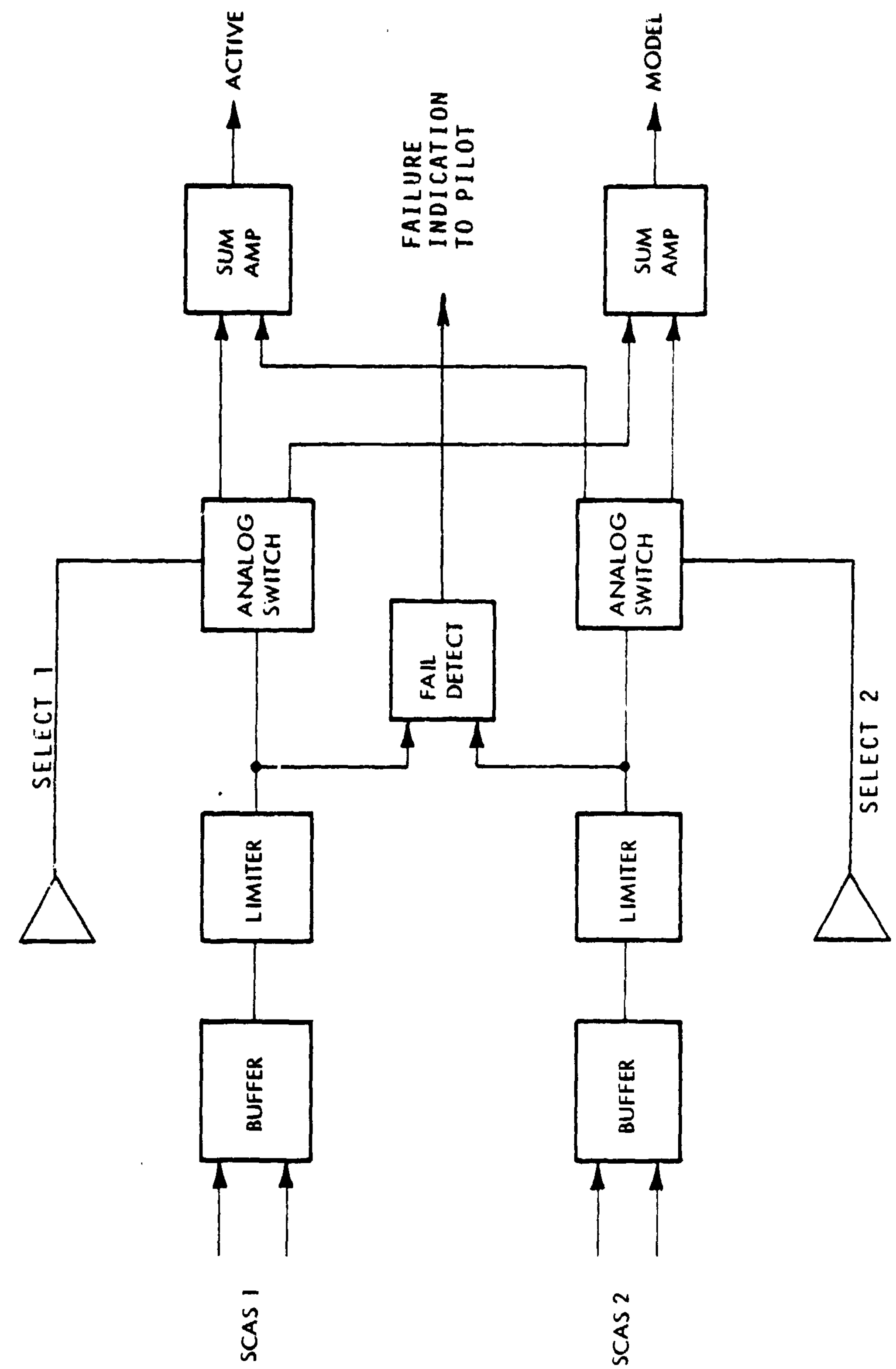

FIGURE 2.28. GUST ALLEVIATION INTERFACE

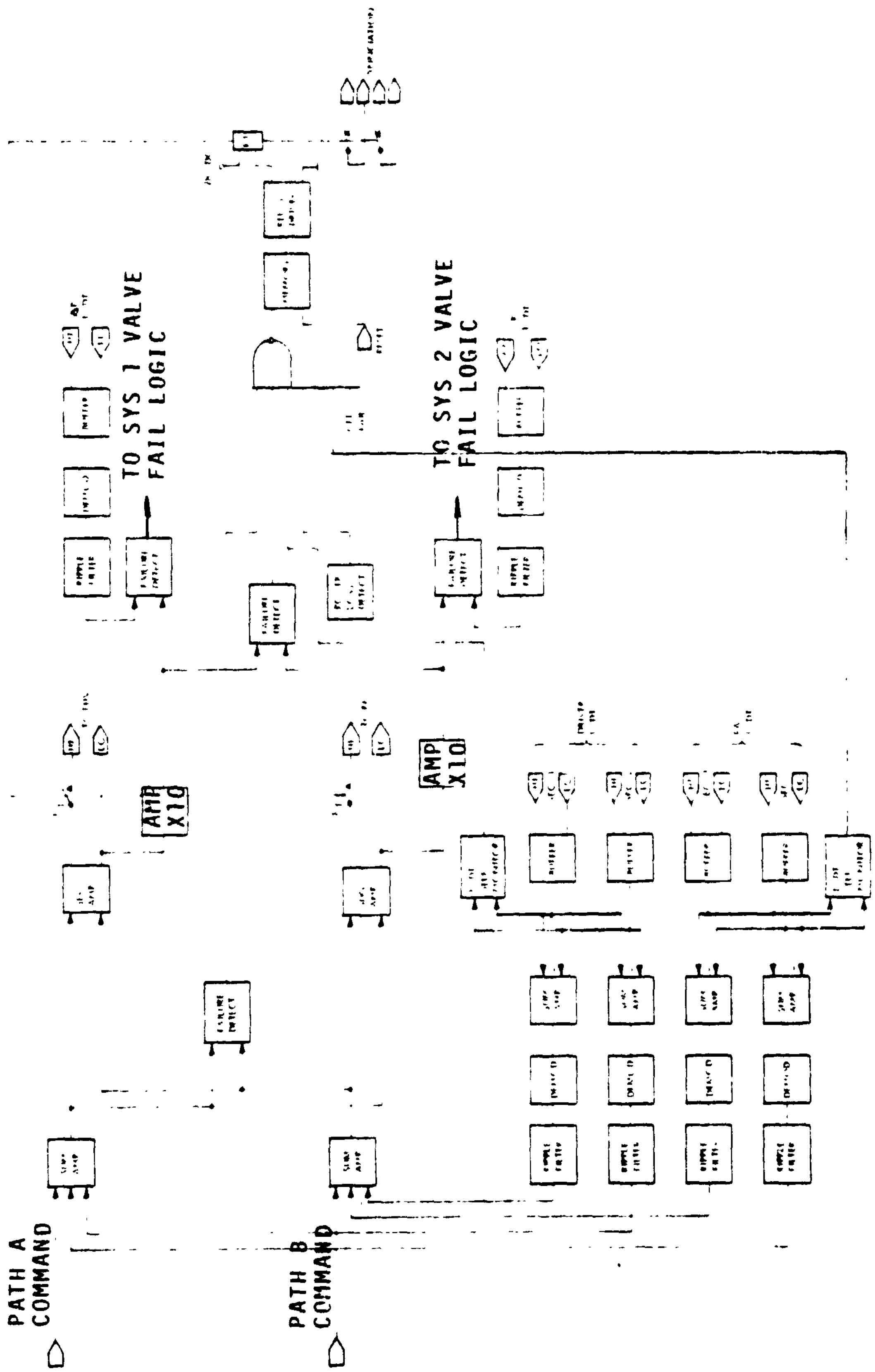

FIGURE 2.29. HYDRAULIC SERVOLOOP AND FAILURE DETECT'

because these are flight-safety critical. Even though the probability of single stage EHV failure is very low (.02 per million hours), steps have been implemented to further reduce the probability of fluid filtering, orifice sizing and immediate shutdown of the actuator section in the event of valve hardovers.

Each control stage has a differential pressure sensor which provides an input to the valve monitor curcuit. The valve monitor circuit senses correct relationship of current versus differential pressure. If current exceeds a certain threshold, pressure must exceed a threshold in the proper polarity. Each EHV is monitored by all three direct electrical link channels. Each channel detects an EHV failure independently; its logic actuates a relay. The three channel relay contacts are wired in series to operate the hydraulic system shutoff valve. This arrangement ensures there will be a low probability of nuisance shutdowns since all three channels must detect an EHV failure to cause hydraulic system shutdown. The only single failure resulting in a nuisance shutdown would be a mechanical open or jam in the differential pressure sensor, the probability of which is extremely low. Electrohydraulic valve hardover failure rate is also extremely low, so the reliability of the monitor in functioning when required need not be great. Passive failures of the EHV are also detected by comparing EHV input currents and the resulting differential pressure.

Control and power stage motion transducer electrical malfunctions are detected by monitoring the common-mode secondary voltage. The transducers are designed so that the common-mode voltage is essentailly constant over the stroke. If excitation is lost, or the coil opens, the voltage will decrease; if a secondary shorts to power, the output will increase. Either condition will cause the monitor to trip. Mechanical failures of the transducers are detected by a periodic check of channel tracking during built-in-test.

2.3.3.4 Major Component Characteristics

Characteristics of the DEL Control Unit Flight Control Actuators, PFCS Panel Maintenance Unit and Control Position Transducers are given in Volume II, Appendix I. Flight Control Actuators were proposed by Bertea System Division of Parker Hannifin Corporation.

2.3.3.5 System Operation/Maintenance

Operation of the system consists of a preflight routine by the pilot and periodic operational check by the crew chief.

Pilot's Operational Check

Before each flight, the pilot boxes the controls on one primary and the secondary flight control hydraulic systems. After run-up he checks operation with the second primary system. During the operation, he monitors for any failures on the Caution/

Advisory panel and the PFCS panel. This check is the same as now performed with the mechanical system.

Periodic Operational Check

At intervals of approximately 10 hours, the crew chief performs a system self-check (Built-in-Test). This routine replaces the periodic inspection of mechanical control runs. The maintenance check consists of running individual channel checks and a Channel Tracking Test. The test consists of initiating test by depressing the "Push to Test" switch on the System Maintenance Unit located in the avionics bay. In approximately two minutes the three channels are checked through an automatic sequence for proper response to simulated failures. The channel test assures that failure detection circuits are not failed passively. At that time the TRACK legend is lighted, the crew chief then boxes the controls with collective pitch down, and again with collective pitch up. This process checks channel tracking. This check uncovers remote failure conditions, such as mechanical failure of an actuator LVDT probe, which are not detected by normal failure detection circuits and mistracking between channels, that might be due to rigging errors. After completion of all required motions, the TRACK legend goes out and the GO legend comes on. This completes the test.

Maintenance

Maintenance is required only if an actuator develops a leak, a failure is detected, or the system fails the operational check.

Flight control system status is displayed on the central caution/advisory panel, on the PFCS panel, and the System Maintenance Panel.

The caution/advisory panel displays the following legends relating to the flight control system.

Flight Control First Fail -- This advisory light flashes for any first failure in a channel; a subsequent first failure causes the flashing to resume. The pilot stops the flashing by pressing the master caution indicator.

Flight Control Second Fail -- This advisory light illuminates for a second failure of a like element. The pilot aborts and lands as soon as possible.

Jam No. 1 and No. 2 -- This advisory light indicates high friction or jam in the corresponding actuator control stage/power valve. The pilot aborts and lands as soon as possible.

Hydraulic System Caution/Advisory -- This is the same as is now supplied for the mechanical system.

SCAS Failure Indication -- This is the same as currently provided in the mechanical system.

Troubleshooting is accomplished with a minimum of special support equipment. The approach is to indicate the probable location of the failure using the fault isolate function of the Maintenance Unit. At this time, the failure may either be definitely isolated to the DEL Control Unit, or it may be in the DEL Control Unit wiring or a component.

To isolate failures of a DEL Control Unit, two of the units would be exchanged and the failure induced. To isolate failures in an actuator, short non-keyed jumpers would be provided to reverse actuator cabling. Failures in a CPT would be isolated by replacing the channel transducers with a dummy plug to check wiring continuity. If a failure in the wiring was indicated, the affected cable or junction box would be replaced. With the multi-conductor cable system proposed, cable failures have a very low probability.

A Maintenance Test Set, similar to the one developed for HLH, interfacing with the DEL Control Unit test connector would be used for rigging of control position transducers and for checking interchannel tracking.

Analysis for the system failure modes show that over 90% of the failures will be in the power supplies or the DEL Control Unit. These failures are easily detectable without use of even the simple methods described here. Experience accrued on the HLH system attested to this fact. The only failures not isolated to the control unit were actuator leaks and one misrigged CPT.

2.3.3.6 Stability and Control Augmentation System (SCAS)

Specifications of the SCAS are given in Appendix III of this volume. Mechanization of the SCAS functions is given in Volume II, Appendix I.

2.3.3.7 Installation

Figures 2.30 and 2.31 show a possible equipment installation arrangement within the XV-15.

Cockpit Area - Mechanical control runs are disconnected and pilot's controls linked to multi-redundant LVDT packages. Existing force feel and trim provisions are retained. The

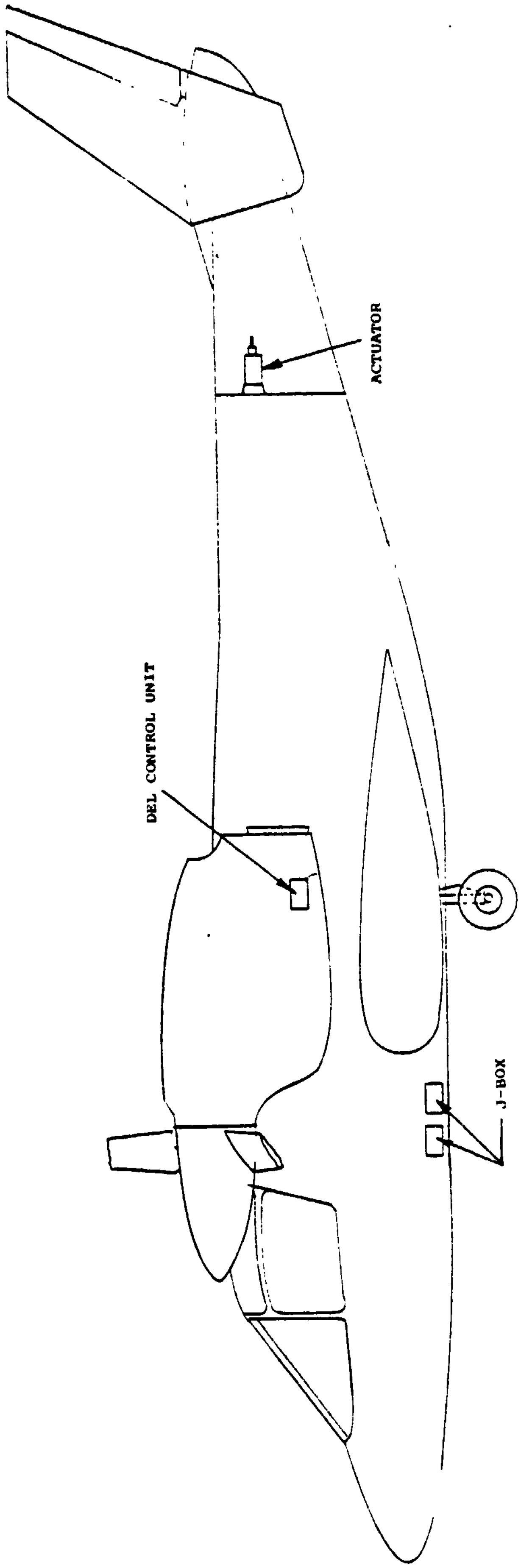

FIGURE 2.30. XV-15 INBOARD PROFILE - SIDE VIEW

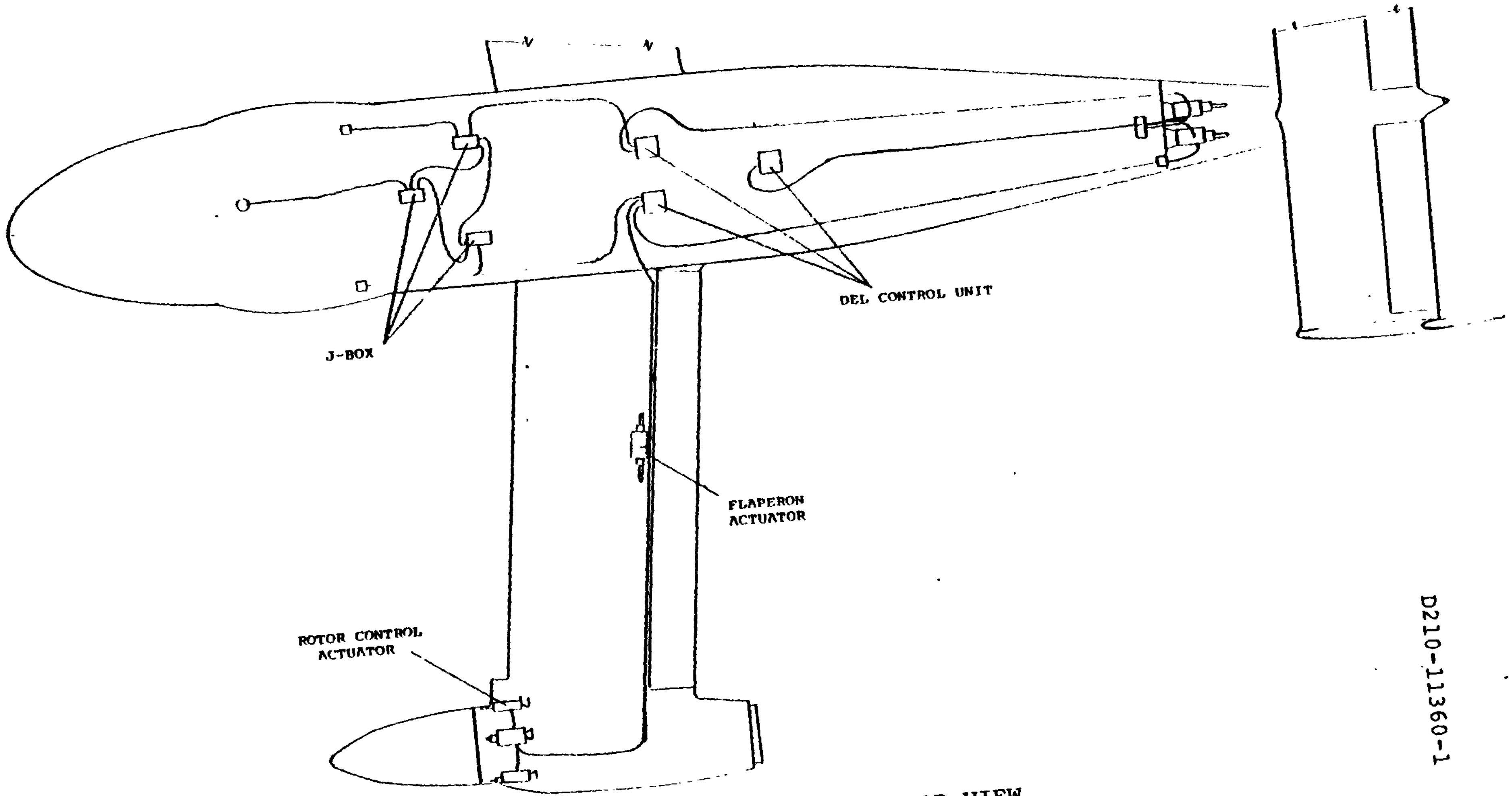

FIGURE 2.31. XV-15 INBOARD PROFILE - TOP VIEW

PFCS panel is located in the center instrument console. An electrical input engine control quadrant replaces the existing mechanical input quadrant.

Avionics Bay - New SCAS Units are installed to replace existing equipment and Maintenance Unit is installed.

Under Cabin Floor - Junction boxes are installed to collect signals from cockpit/avionics bay for transmittal to the DEL Control Units.

Control Unit Installation - The units may be installed wherever a suitable space is available. The space aft of the wing, presently housing the mechanical mixer is a candidate.

Actuators - Details on rotor control installation are shown in Figure 2.2. Airplane surface actuators are mounted to replace the existing actuators. New mounting and revised plumbing are required.

Details on proposed system cabling are shown in Figure 2.32. All cables would be shielded multi-conductor units terminaled in self-locking threaded connector series MIL-C-83723. Shielding is included as a protection against lightning effects; analysis and test of system interface circuits indicates it is probably not required. It was not used on HLH ATC; however, the 347 aircraft was not cleared to fly near thunderstorms.

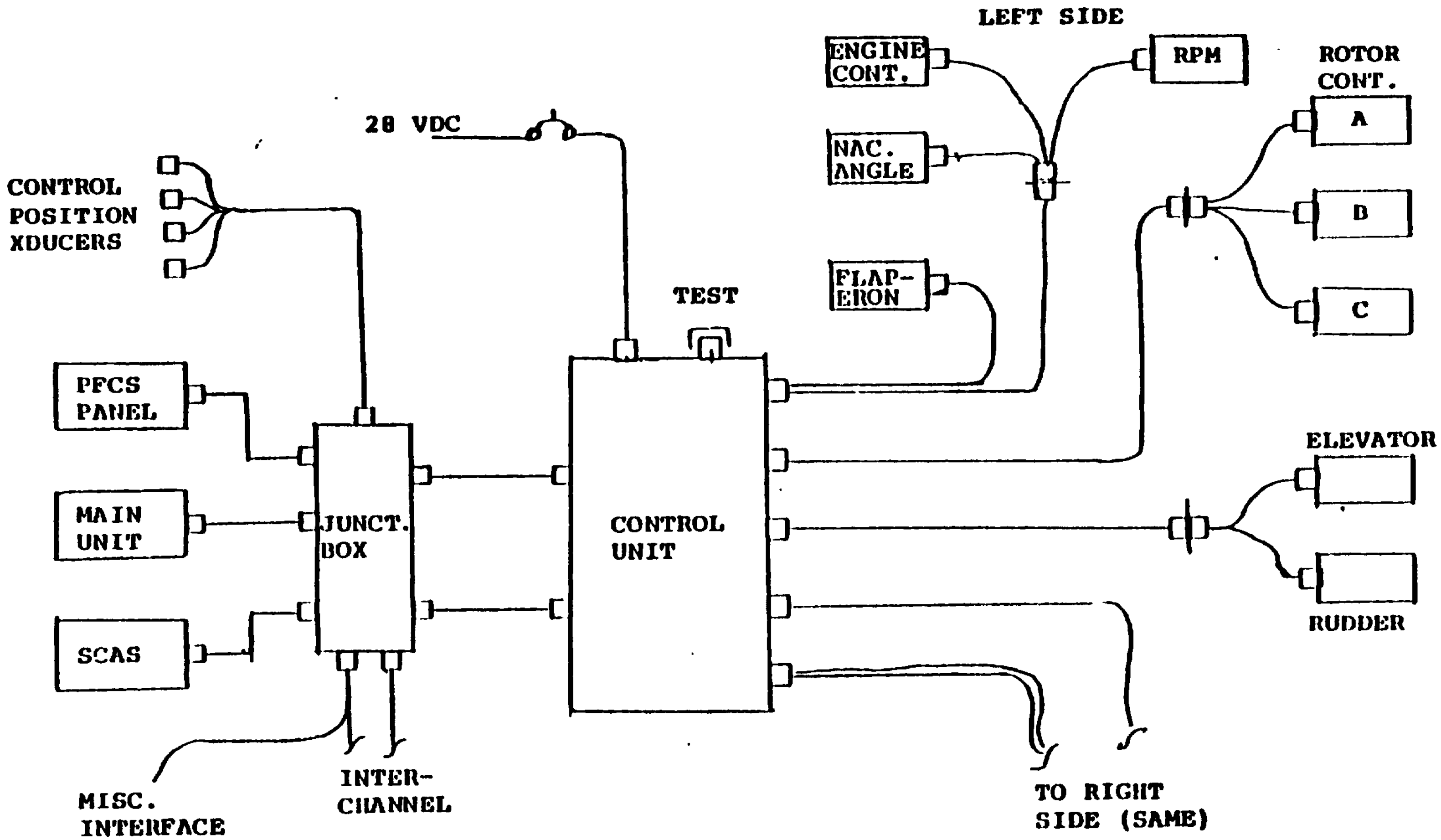

FIGURE 2.32. PRIMARY FLIGHT CONTROL SINGLE CHANNEL INTERCONNECT

2.4 POWERPLANT INSTALLATION

The powerplant installation is shown in Drawing SK-27244 (Figure 2.2) and schematically in Drawing SK-27256 (Figure 2.33).

2.4.1 Engine

The Lycoming LTC1K-4K direct-drive version of the T53 engine per Reference (2) is utilized with no changes from those on the current XV-15 aircraft. The "patch-on" engine gearbox (Figure 2.9) is used to assure engine commonality with basic XV-15.

2.4.2 Mounting

The engine is mounted fixed (non-tilting) to the structural adapter described in Section 2.2.2 and Drawing SK-27245 (Figure 2.5). Rationale for the fixed versus tilting engine is found in Appendix I. A three-point mount system is employed using machined aluminum fittings as shown in Figure 2.2. Two of the three forward mounting pads on the engine inlet housing casting are picked up - at the top and lower inboard locations, and one of the aft pads (lower inboard) on the periphery of the diffuser housing is used. The forward inboard mount takes loads in all three directions; the forward top mount resists only lateral loads. The aft mount takes vertical and lateral loads, but allows engine thermal growth longitudinally without taking load. The forward mount system allows engine radial expansion.

2.4.3 Cowling

The engine cowling consists of forward, center, and aft sections. A schematic sketch of the cowl is shown in Figure 2.34.

a) Forward Section - This section consists of an outer cowl annulus around the air inlet, and a central bullet covering the engine gearbox. Three airfoiled struts, one of which encloses the drive cross-shaft, connect the inner bullet and outer annulus; both parts are mounted to the engine, and are removed in the forward direction. They are constructed of built-up aluminum pieces.

   The bullet fairing nose section forward of the split line at the cross-shaft centerline is supported by a bulkhead mounted on the gearbox housing. This part, the forward section of the inlet struts, and the outer cowl lip section are tied and removed together to expose engine gearbox and cross-shaft. The portions behind the split line of bullet, inlet airfoil struts, and annulus are also tied together and removed forward after cross-shaft section removal and removal of the nuts on the engine mounting face studs which secure this section. The other annulus portion is supported on the engine and around the air inlet.

b) Center Section - This section extends essentially the length of the engine less tail pipe, and is constructed of aluminum rings and stringers. It contains two large removeable access panels each extending the full length of the section and as wide as the engine. These are located at top and bottom, and are attached by quick-release fasteners to the side panels and end-frames which in turn are supported by the structural adapter (airframe-mounted). Removeable panels were selected over hinged access doors to simplify cowl design and reduce costs. Maintenance is done using separate stands; quite acceptable for a research vehicle operating from a large fixed base. No integral workstands are provided in the cowl.

c) Aft Section - The short tail-cone section is attached to and supported by the aft frame of the center section and is removed aft.

2.4.4 Firewalls

A ring fire seal attaching to the engine fire shield is located between center and aft sections of the cowl, separating the engine hot section from items forward. A firewall also separates the engine compartment from the airframe structural adapter. Engine service lines pass through the firewall.

TABLE 2.4. KEY TO FIGURE 2.33

1. LTClk-4K Engine
2. Engine Gearbox (Ref.)
3. Structural Adapter (Ref.)
4. Forward Top Engine Mount
5. Forward Inboard Engine Mount
6. Aft Engine Mount
7. Cowl Forward Section Outer Annulus
8. Cross-Shaft Fairing Strut
9. Cowl Forward Section Inner Bullet Nose Section
10. Cowl Forward Section Inner Bullet Aft Portion
11. Engine Mounting Face Studs
12. Cowl Center Section Forward Ring
13. Cowl Center Section Upper Access Panel
14. Cowl Center Section Lower Access Panel
15. Cowl Center Section Fixed Outer Panel
16. Cowl Center Section - Structural Adapter Attach Section - Forward
17. Cowl Center Section - Structural Adapter Attach Section - Aft
18. Cowl Aft Section
19. Ring Fire Seal
20. Firewall
21. Cowl Air Inlet
22. Engine Air Inlet
23. Optional Inlet F.O.D. Screen
24. Exhaust Pipe
25. Engine Cooling Air Scoop
26. Engine Compartment Cooling Air Inlet
27. Engine Compartment Cooling Air Exit Screen
28. Engine Starter-Generator
29. Gas Generator Control Actuator and Linkage
30. Engine Oil Tank
31. Engine Fuel Feed Line
32. Firewall Fuel Shut-off Valve
33. Fire Detection Wire (Partial)
34. Fire Detection Junction Box
35. Fire Extinguisher Agent Bottles
36. Air Bleed Valve (Ref.)
37. Air Bleed Duct (Ref.)

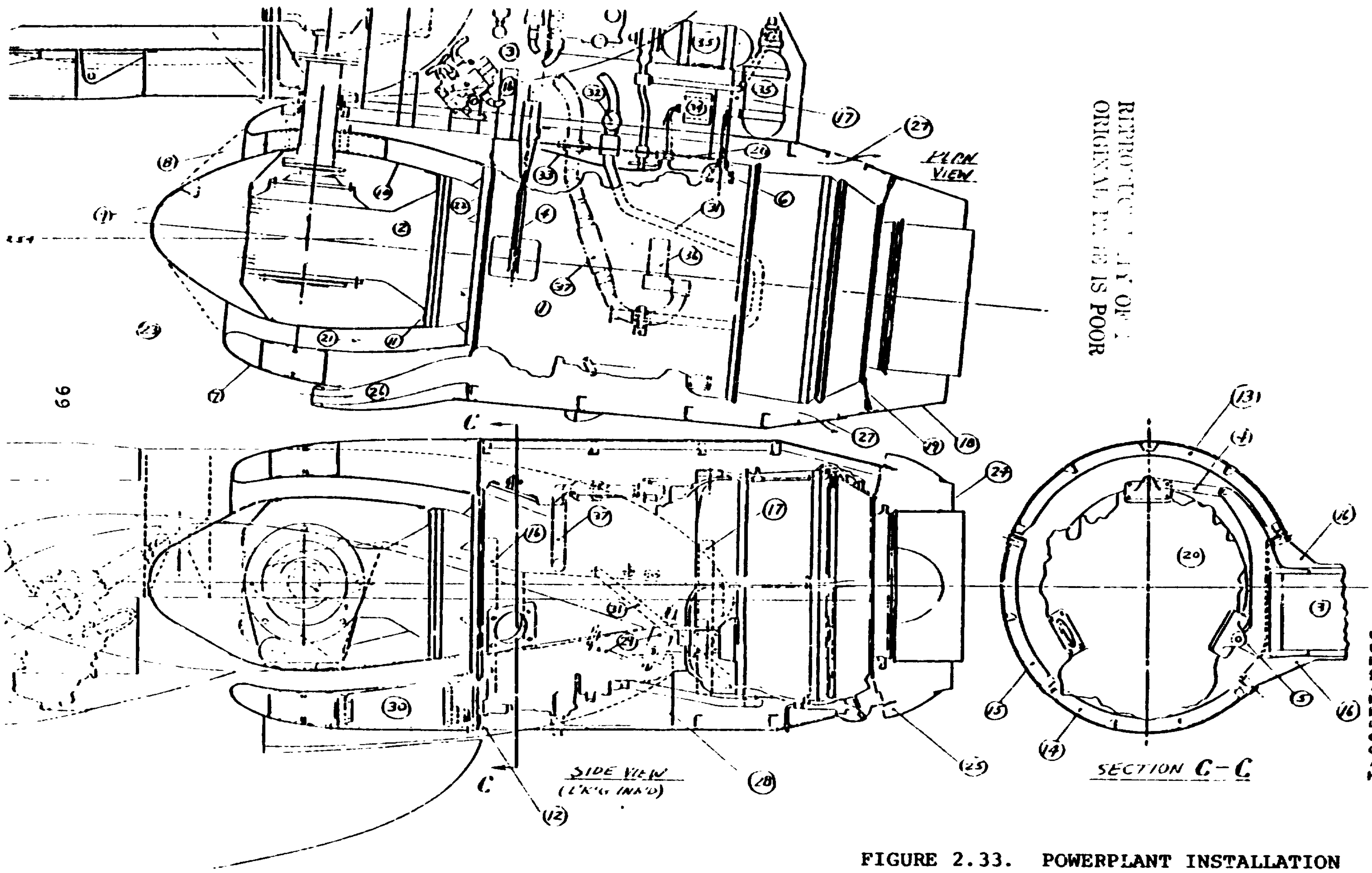

FIGURE 2.33. POWERPLANT INSTALLATION

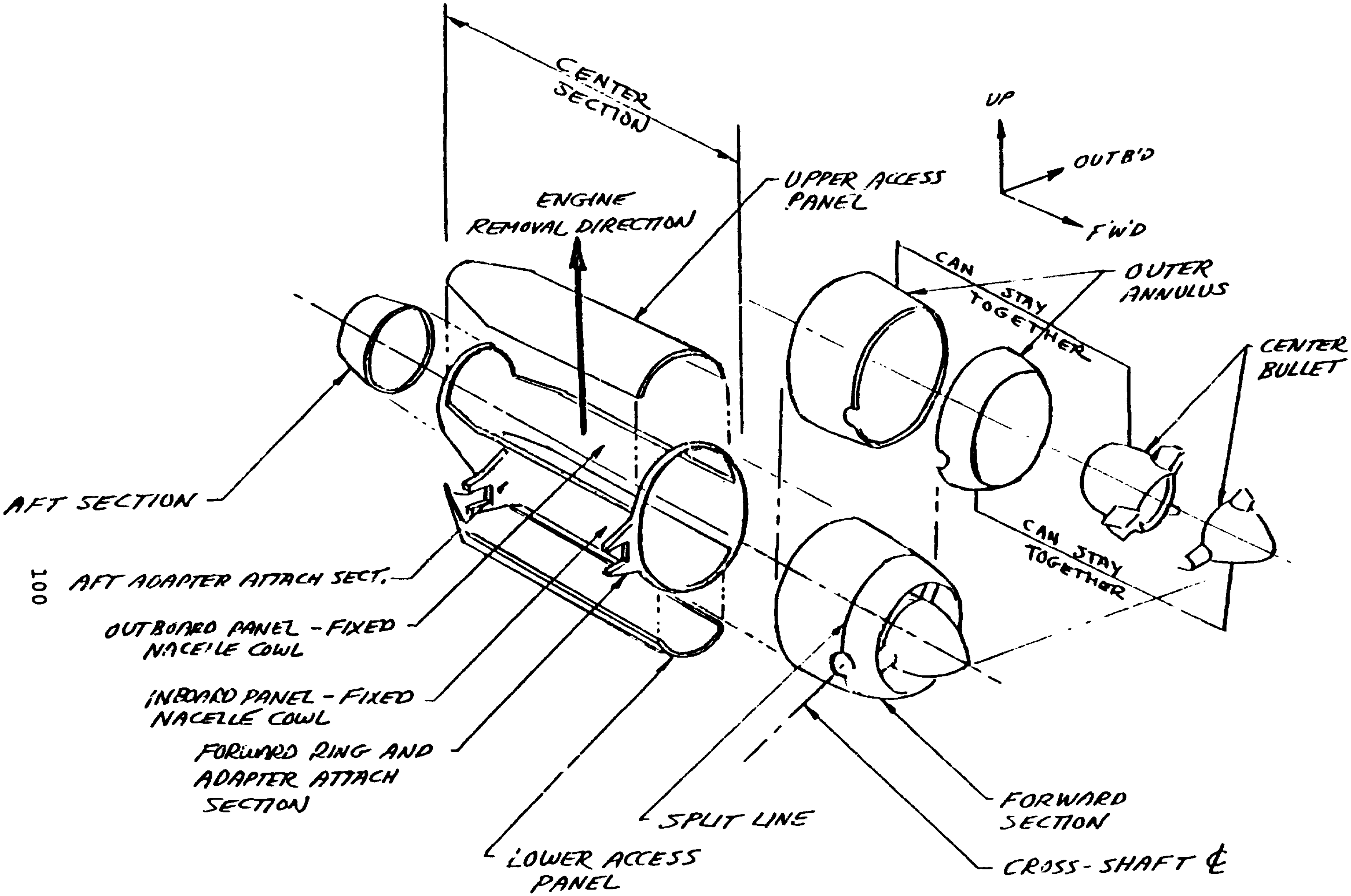

FIGURE 2.34 ARRANGEMENT OF COWLING - LEFT SIDE ENGINE NACELLE

2.4.5 Engine Air Induction

The cowl air inlet is an annulus to match the engine inlet with sections designed as a fixed geometry system in a compromise between hover and cruise flight conditions. Three radial airfoiled struts run between the centerbody (bullet) and the outer cowl, one of which encloses the cross-shaft drive from engine to intermediate transmission. An inlet FOD screen can be utilized with this system if desired.

2.4.6 Engine Air Exhaust

A short straight pipe is attached to the engine exhaust flange via a clamp. The pipe is sized as a compromise between hover and cruise flight to provide zero jet thrust at 300 knots speed.

2.4.7 Engine and Engine Compartment Cooling

a) Engine - Cooling air must be provided for the rear face of the engine power turbine assembly. Two air scoops are provided in the skin of the aft cowl section just behind the ring fire seal to provide a source of cooling air to be sucked through the exhaust diffuser support struts and turbine bearing housing and across the aft face of the power turbine into the exhaust stream.

b) Engine Compartment - Cooling air is taken into the engine compartment from a ram air inlet on the outboard side of the cowl forward section. The air exists just forward of the ring fire seal via screened openings.

The DC starter-generator mounted on the engine is integrally cooled. If, in the detail design and analysis phase, it is found that this concept does not provide sufficient hover flight cooling for certain compartment-mounted components, the alternatives of either integrally cooling these components or of changing to exhaust ejector cooling by changing the ring fire seal to a porous fire screen will be investigated.

2.4.8 Starting

The starting system is electrical and includes, as does the current XV-15 aircraft, a 300-ampere self-cooled DC starter-generator mounted on the engine accessory pad on the lower side of the engine.

2.4.9 Controls

Control of the engine is integrated with the electrical fly-by-wire flight control system of the HTR XV-15 aircraft.

Control of the gas generator is the primary input. An engine-mounted redundant electromechanical actuator drives the power lever splined shaft on the fuel control via linkage. The power turbine speed selector lever on the engine-fuel control will be set in fixed position for overspeed (topping) control only, and no actuator is required. The power turbine will be under control of the fly-by-wire thrust management system.

2.4.10 Lubrication

This system is self-contained within the engine except for an airframe-supplied oil tank. It is completely separate from any transmission lubrication system. The oil tank (approximately one-gallon capacity) is located in the lower area of the cowling forward section outer annulus. As on the current XV-15, oil cooling will be provided by a fuel-oil heat exchanger in the fuel feed line downstream of the fuel control, and supplied by the engine manufacturer.

2.4.11 Fuel System

Except for a small additional length of feed line and the elimination of the swivel joint required for a tilting engine, the fuel system of the current aircraft will be preserved on the airframe side of the firewall. On the engine compartment side, system components will be similar or identical. These include fuel filter, fuel/oil heat exchanger, and flowmeter (the latter is airframe supplied).

2.4.12 Fire Detection

A continuous, re-setting, wire loop-type detector system is provided in each nacelle routed around the engine compartment with a junction box located on the airframe side of the firewall. The system is similar to that on the current aircraft and will be designed to match cockpit displays and controls.

2.4.13 Fire Extinguishing

A two-shot $CBrF_3$ extinguishant system is provided for each nacelle. The two agent bottles are mounted on the aft section of the fixed structural adapter just inboard of the firewall. Lines run from the bottles through valves and across the firewall to distribution nozzles in the engine compartment. The system is similar to that on the current aircraft and will be designed to match cockpit controls. Bottles are located for easy inspection and removal.

2.4.14 Ice Protection

Engine inlet anti-icing is supplied by the engine manufacturer. It can be operated manually by a switch in the cockpit which electrically de-energizes the engine hot air solenoid valve. If automatic operation is desired, an icing detector could be included in the system. (Icing detection is mentioned in Reference 2 on page 98. The forward section of the airframe cowling around the primary air inlet is not ice-protected. A hot air duct and/or electric blanket system could be added if warranted; however, it is not believed necessary in a research aircraft normally flown in a benign environment.

2.4.15 Air Bleed

As in the current XV-15 aircraft, engine bleed air for the aircraft environmental control system is taken from the engine in the right side nacelle.

A stainless steel bleed air duct is provided from an electrically actuated valve (airframe supplied) mounted on the engine bleed air port. The duct is insulated, has a flow limiter incorporated, and includes couplings to allow for thermal expansion. The duct runs through the firewall and connects to the bleed air line in the right wing. No swivel joint is required with the fixed engine. In the current XV-15, a pre-cooler (heat exchanger) is employed to reduce air temperature to 275°F. in the line through the wing and the fuselage to the ECU, and a blower driven by a utility system hydraulic motor provides the cooling air needed for the low temperature side of the cooler in hover flight. A similar system can be provided in the HTR aircraft. Hydraulic motor, blower, and heat exchanger would be mounted under the structural adapter and within the contour of the fixed aft fairing. A less expensive alternative to be evaluated in a detail design phase is to omit the pre-cooler and pass the bleed air at the higher temperature through wing and fuselage ducting with (additional) insulation added and on to the ECU. More details of the current system would have to be known.

The preliminary design evaluation of the powerplant installation using the fixed engine arrangement indicates that it is simple and straightforward. Access for inspection, maintenance, and engine changes should not pose problems. All subsystems are conventional and should match up well with the current aircraft. The subsystems are simplified by absence of requirements for service line swivel joints. Handed parts are minimized in the design; for instance, the cowl access doors are interchangeable side to side.

2.5 SUBSYSTEMS

A schematic arrangement of the subsystems and service lines is shown in Drawing SK-27256 (Figure 2.35).

2.5.1 Hydraulics System

The current XV-15 hydraulic system requires some changes to be made to accommodate the hingeless tilt rotor and Boeing Vertol control system design philosophy. Figure 2.36 shows a simplified schematic of the modified system. The changes are confined, however, to the nacelle area. Current XV-15 components can be used inboard of the wingtips where all the hydraulic loads, with the exception of the rotor power actuators, and all the fill points are located. In the revised arrangement all rotor actuators can be served by all three hydraulic systems - the two dedicated to flight controls and the backup utility system which can be utilized via a switching valve system. Hydraulic pumps of a design employed on the Boeing Vertol YUH-61A prototype aircraft are used mounted on the accessory gearboxes (AGB) in the tilting nacelles as shown in Drawings SK-27244, SK-27256, and SK-27251. The pumps are integrally air cooled. A primary flight control pump is located in each nacelle and the utility system pump is on the left nacelle AGB. Eight hydraulic lines per nacelle are brought across the tilt joint from rotor nacelle to fixed afterbody. As noted previously, properly guided and constrained slack loops in the lines will provide for operation of the rotor nacelle through a 95° maximum angle. Other

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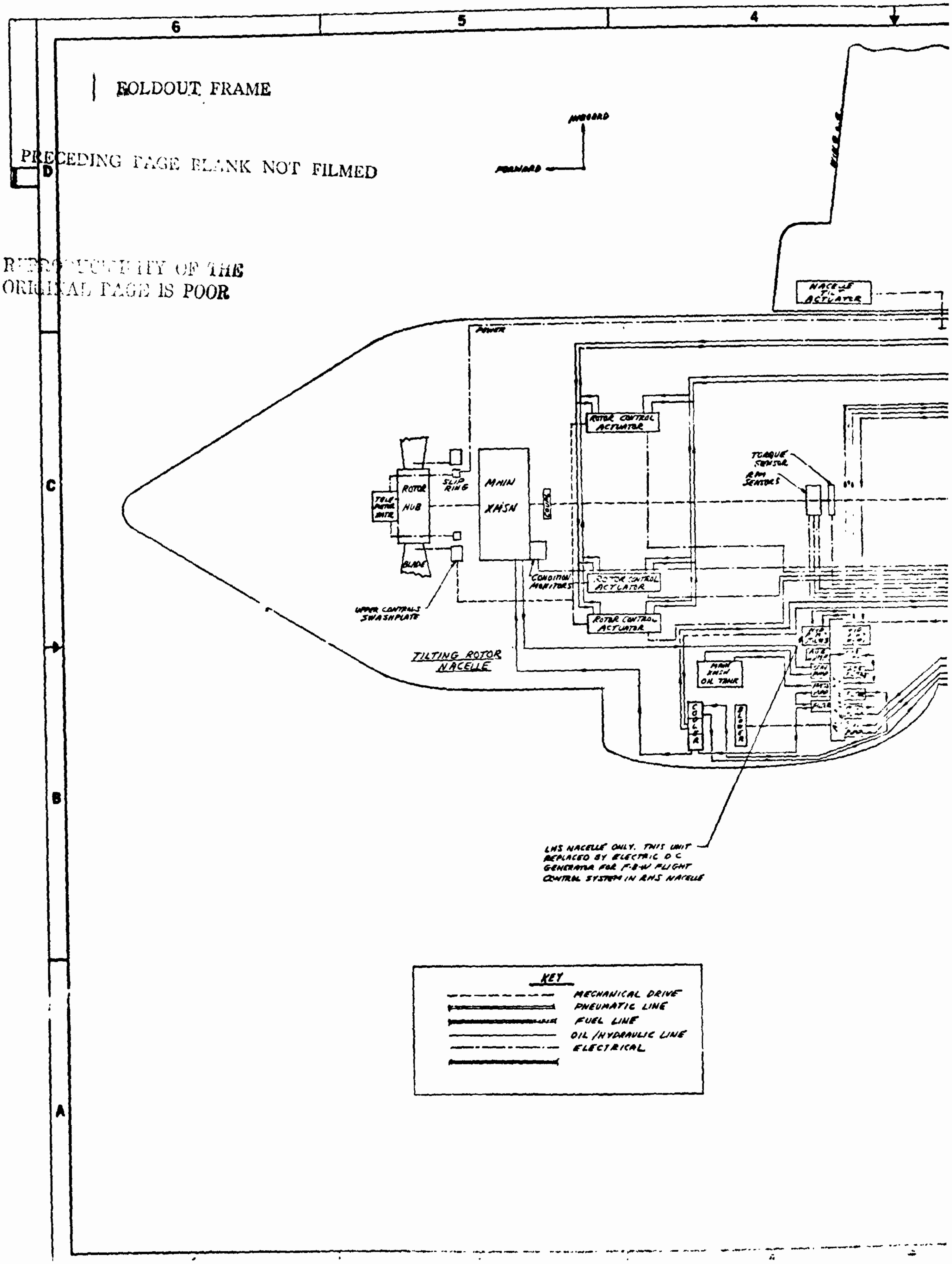

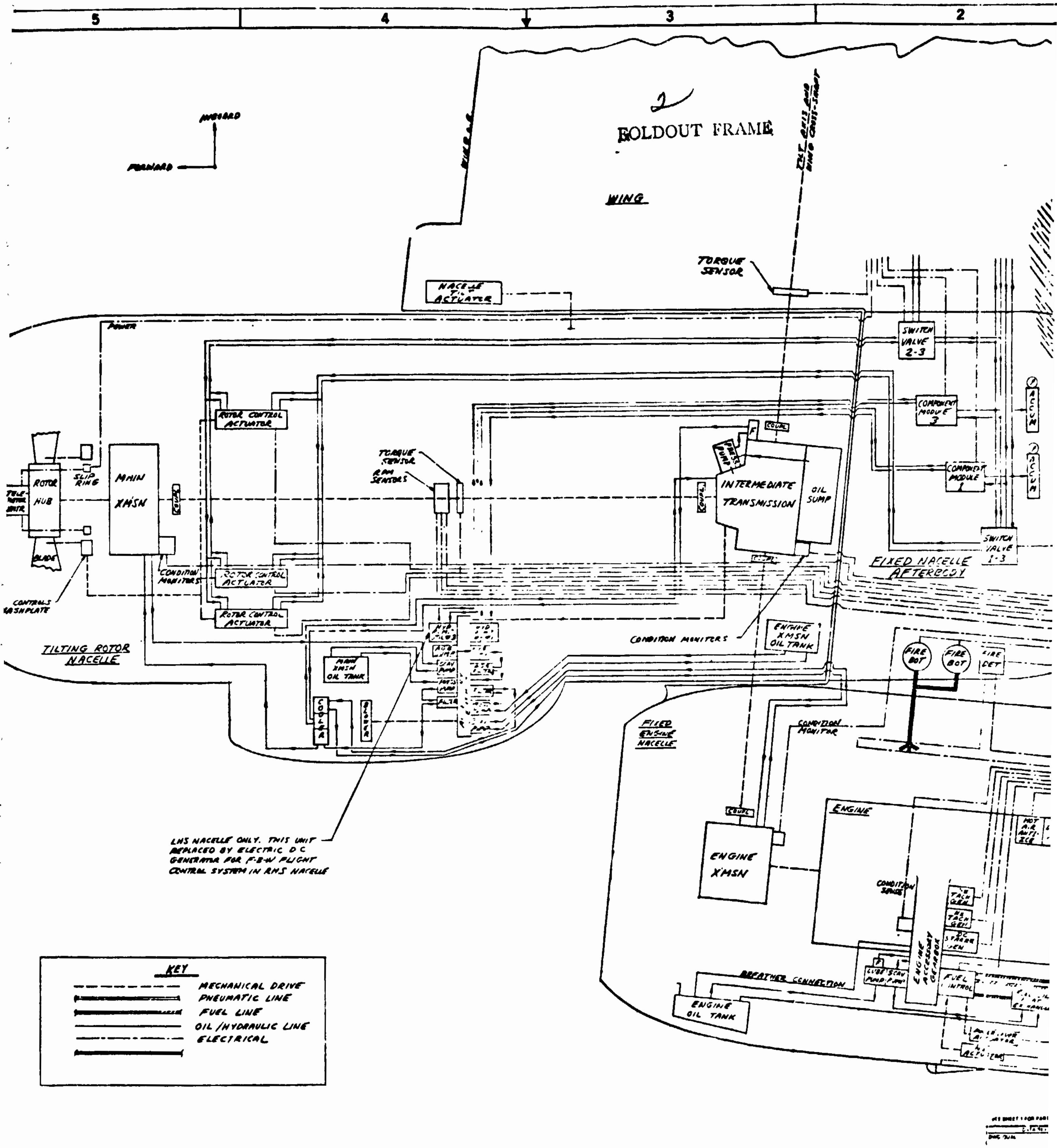

FIGURE 2.35. NACELLE SCHEMAT
HTR XV-15 - (LH

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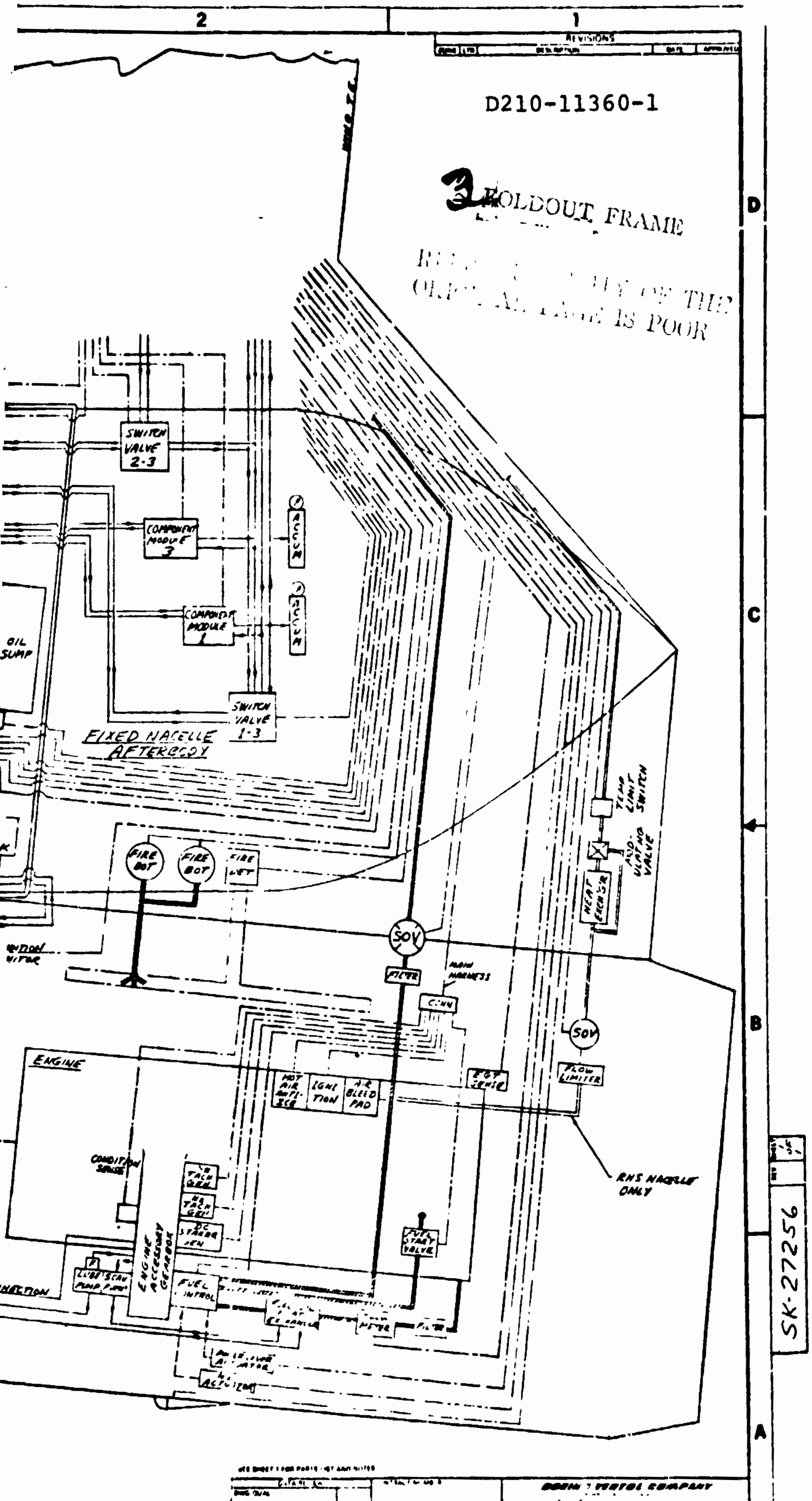

.35. NACELLE SCHEMATIC - SUBSYSTEMS COMPONENTS - HTR XV-15 - (LHS NACELLE)

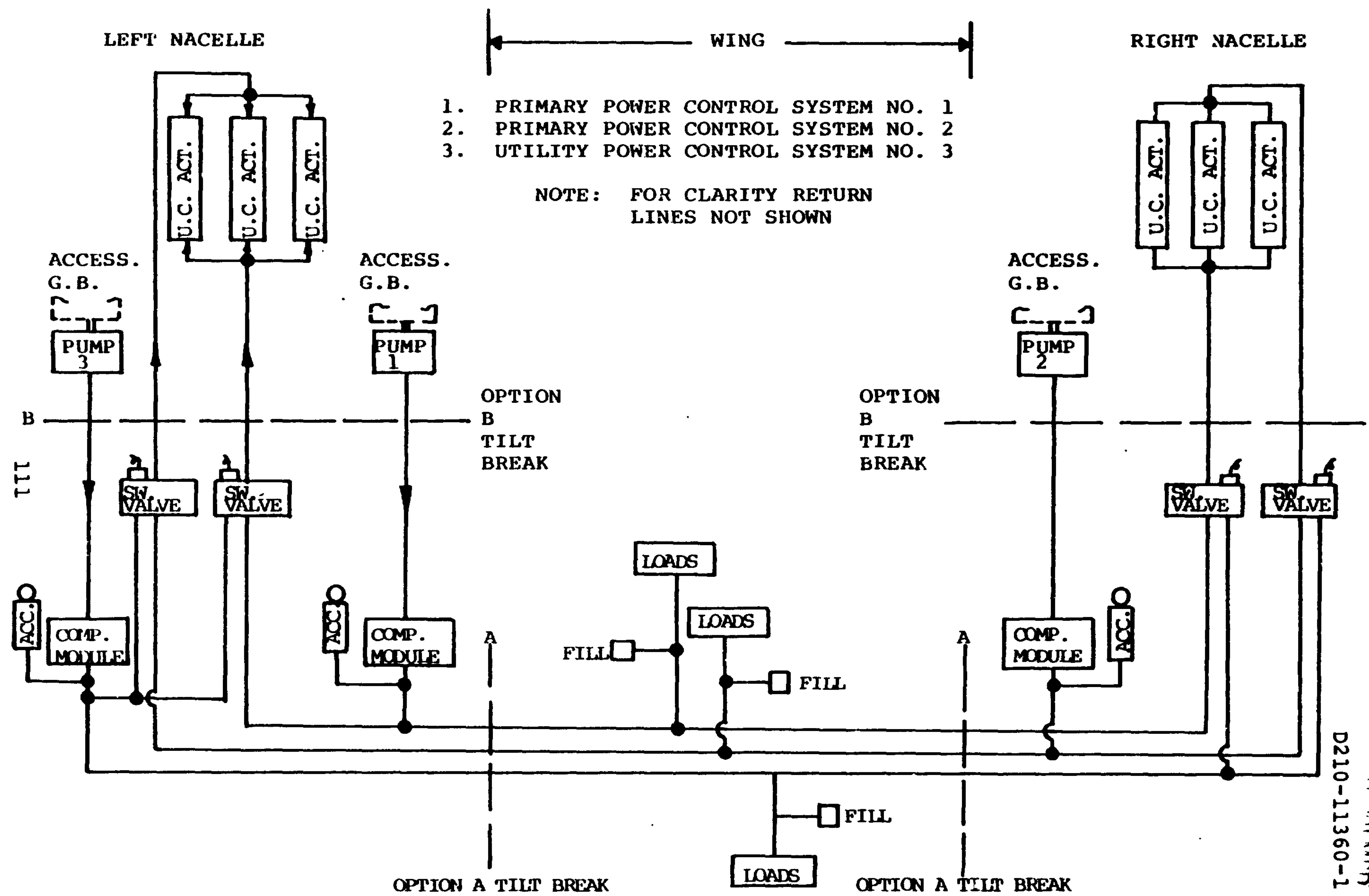

FIGURE 2.36. SIMPLIFIED SCHEMATIC - XV-15 HYDRAULIC SYSTEMS MODIFIED FOR BOEING VERTOL ROTOR AND FIXED ENGINE NACELLE

components of the hydraulic power supply system are located in the fixed afterbody section of the rotor nacelle. The left side nacelle contains two each component modules, accumulators, and switching valves. These units are mounted on top of the fixed adapter structure as shown in Drawing SK-27244, and are thus readily accessible for inspection and maintenance via access panels in the afterbody. All lines are run outside the structure of the nacelle. Component modules and other components are similar to those used in the Boeing Vertol YUH-61A prototype aircraft. The component module concept simplifies the system and reduces potential leak points by combining such items as reservoirs, relief valves, thermal and pressure sensors, filters, and other valves and items necessary to the system.

2.5.2 Electrical System

On the XV-15, the electrical generation, conversion, and distribution systems are arranged so only two components are located in the nacelles - one 30 V.D.C. 300-ampere starter generator pad mounted low on each engine. These units provide engine starting acting as battery-powered motors and then act as generators feeding the two DC bus systems. Generator controls, the two batteries, DC and AC busses, DC to AC converters (2), and associated relays are all mounted in the aircraft fuselage and electrical cabling connects across the wing from nacelle generators to the fuselage items noted.

On the Boeing Vertol version of the aircraft, there are two basic changes in the system. The fly-by-wire flight control system requires triple electric supplies - thus, a small DC generator is added in the right side rotor nacelle driven by the accessory gearbox to give a third basic supply to the two already aboard. And since the engine in the Boeing Vertol nacelle does not tilt like that in the basic XV-15, only one generator output cable (that of the added item) runs across a tilt joint - on the right side - instead of on both sides from the engine-mounted generators on the current aircraft. There are various electrical loads on both the tilting and non-tilting positions of the Boeing Vertol nacelle; some of these are:

Loads - Fixed Nacelle	Loads - Tilting Nacelle
Engine Transmission Sensors	Rotor & Intermediate Xmsn Sensors
Engine Controls (F-B-W)	Upper Controls Actuators (F-B-W)
Engine Condition Sensors	Rotor Telemetry Xmtr Power
Engine Starter	Hyd. Switch Valve Controls (possible)
Fuel Flowmeter Sensor	Hyd. System Sensors (possible)
Fire Detectors	RPM/Governor Sensors
Extinguisher Sys. Controls	Torquemeter Output
Nacelle Nav. Lights	
Bleed Air Valve Control	

2.5.3 Pneumatic System

The engine air bleed system is the only pneumatic system in the aircraft and is described in Section 2.4.15.

2.5.4 Cockpit Displays/Instruments

A review has been made of cockpit display and control items to assess potential changes required because of an installation of Boeing Vertol HTR nacelles on the XV-15 aircraft. When more data on the current XV-15 is available a more detailed evaluation can be made.

Pilot and Copilot Instrument Panel Displays

- Remove rotor flapping indicator.
- Critical monitor meter is interpreted as a cruise guide indicator.

Instrument Panel Center Section

- Add four transmission oil temperatures and pressure indicators.
- Add a voltmeter and an ammeter for the third channel electrical supply.
- Revise 40-segment caution panel if and as required for fly-by-wire control system caution information, and to include space for total of six chip detectors, six pressures, and six temperatures to indicate drive system condition.

Center Console, Left Side

- Add a new primary flight control system panel.
- Possible changes are anticipated in the stability and control augmentation panel, the RPM governor panel, and the two throttles due to the new flight control system.

Center Console, Right Side

- Changes are anticipated in the RPM governor control wheel and indicator.

Overhead Console

- Changes are anticipated in the electrical system control panel and the electrical system circuit breakers.

2.5.5 Aircraft Test Instrumentation

The approach taken in evaluating the work required to provide adequate instrumentation for the flight test program assumed that the aircraft instrumentation used for prior flight evaluations exists and was adequate. Thus, only those items which are specific to the nacelle and pertinent to the testing are discussed here.

One major change from previous practice results from the design. The rotating system data are not extracted via sliprings, but are transmitted to the fixed system using an Acurex-Autodata Wireless Measurement System to provide a repatchable, four channel system from each rotor.

- Acurex-Autodata Wireless Measurement System.
  Static Strain, Bandpass DC to 1 KHz, Four-Channel Capability, consisting of the following:
  1. Model 155S-4 four-channel receiver. One each required.
  2. Model 106S static strain signal conditioning cards. Four each required.
  3. Model 206A static strain transmitter. Four each required.
  4. Model 230A induced power regulator module. One each required.
  5. Model 234A custom induced power matching coil. One each required.
  6. Model 160A induction power oscillator. One each required.
  7. Model 160PS power supply. One each required.
  8. Model 1211S custom antenna matching network. One each required.

Rotating system basic instrumentation is anticipated to consist of blade root bending about two orthogonal axes for each blade and pitch link instrumentation to provide blade root torsion. The hub varrels will be instrumented for bending. Rotor shaft instrumentation will consist of bending bridges and torque gages.

The control system instrumentation is intrinsic to the system itself. Cockpit control and actuator positions can be recorded from the fly-by-wire control transducers suitably buffered to prevent interference with system operation. The actuator force data will be recorded by strain gages on the lugs or shaft.

Six accelerometers will be required on each nacelle frame arranged to measure the six components of linear and angular acceleration. Five additional accelerometers will be needed for each engine installation and at least one on each gearbox in the transmission.

Strain gages will be used to provide strain data and loads on critical components of the flight-load paths.

Normal flight instrumentation is required to measure -

a. Transmission condition (temperature, pressure).

b. Engine condition (temperature, pressure).

c. Engine and rotor/drive tachometers.

d. Engine fuel flow.

e. Electrical system amperage and voltage.

f. Hydraulic system oil temperature and pressure.

g. Caution panel displays.

h. Torquemeter output.

2.6 POWERED SEMI-SPAN TEST STAND IN NASA AMES 40-FOOT BY 80-FOOT WIND TUNNEL

Figure 2.37 shows the HTR XV-15 integrally-powered nacelle assembly vertically mounted on an existing semi-span wing panel which is in turn adapted (using an existing assembly) to the NASA Ames 40-foot by 80-foot wind tunnel floor. The nacelle system, consisting of a tilting (equivalent to lateral swiveling on the test unit) rotor portion, a fixed afterbody fairing, and an engine nacelle enclosing an operable turboshaft engine to drive the rotor, is essentially as shown in Drawing SK-27244, Figure 2.2).

The principal differences from the flight nacelle shown in the above drawings are that the engine and drive lubrication systems are revised, the wing cross-shaft and the accessory gearbox are eliminated (engine-to-rotor drive components are retained), and the tilt/swivel actuator is eliminated. The nacelle has provisions for manual swivel angle adjustments to several positions to duplicate various cases of hover, transition, and cruise flight. The existing semi-span wing tip is modified to accept the nacelle. Nacelle and wing tip modifications all appear feasible, and the combination should provide an acceptable integrated, powered test rig for the HTR XV-15 nacelle in the NASA 40-foot by 80-foot wind tunnel.

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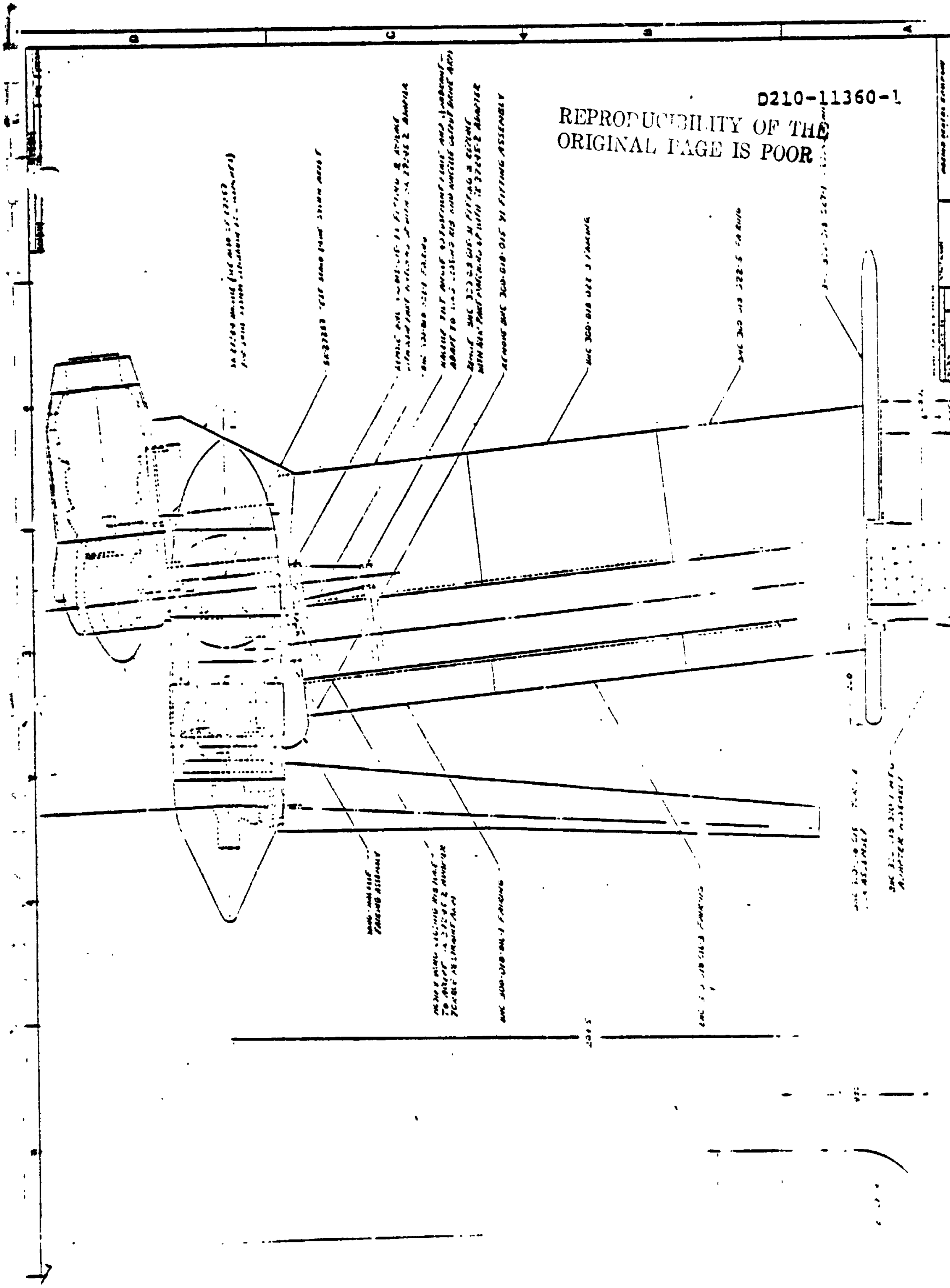

FIGURE 2.37. POWERED NACELLE ON EXISTING WIND TUNNEL TEST - SEMI-SPAN WING - 40- X 80-FOOT TUNNEL - HTR XV-15

2.6.1 NACELLE SUPPORT STRUCTURE AND AIRFRAME

There are three major structural interfaces between the nacelle and the existing semi-span wing panel torque box (BHC 300-018-015-1). The first consists of the two pillow-block fittings aft of the outboard end of the rear spar which form the primary support structure for a tip nacelle. The existing fittings (-31 and -93), even with bushings removed, are too small in bore diameter (55-60%) to accept the spindle insert portion of the SK-27245-2 structural adapter. This adapter was sized to fit into the pillow-block of the actual XV-15 aircraft wing after removal of bearings. Either the fittings must be replaced or the adapter spindle revised. It was considered better to make new torque box fittings rather than modify the adapter, since this is simpler, the joint is stronger, and it is more appropriate to modify something other than a large piece of flight hardware. New fittings of greater bore and O.D. will be made. These larger fittings will make local "bumps" in the wing contour, and the BHC 300-018-22-1 outer trailing edge fairing section will thus require local modification.

The second major interface is the torque reaction tie-in required between the nacelle adapter (which does not tilt/swivel) and the torque box closing tip rib. This tie is required to keep the engine nacelle and aft fairing section fixed. Detail definition cannot be provided on this connection since the adapter-to-XV-15 flight wing tie has not been resolved because insufficient information is available concerning flight wing details. There should be no basic problem bolting a torque reaction arm on the adapter into the tip rib of the existing test wing panel once this information is available.

The third interface is the connection to provide a manually step-variable tilt/swivel angle position of the rotor nacelle with respect to the wing in place of the tilt actuator used on the flight aircraft. This would consist of an adapter fitting mounted on the existing actuator output drive arm (see view C-C of SK-27244) with provisions for through-bolting to a quadrant-plate fixed to the cosing rib of the test wing torque box. This quadrant would be mounted edge-on to the tunnel windstream, and would include multiple bolt holes through a 95° arc to allow for manual swivel angle adjustment. Means for restraining and carefully moving the tilting nacelle during adjustment will be required.

Since the configuration of the HTR XV-15 nacelle differs from those previously used with the test wing, other wing tip modifications are necessary. The BHC 300-018-015-91 fitting assembly at the tip is removed. New wing-to-nacell fairing assemblies are required. The new fairing assembly would consist of a leading edge portion, central section, and trailing edge section on both top and undersides of the wing. All pieces would be easily removable. Additional modifications to the test wing will undoubtedly have to be made to run and tie-in the many and various subsystem lines from the tunnel section exterior up to the nacelle.

The flight nacelle will be provided with hoist hard points not only for lifting and mounting on the XV-15 aircraft in the conventional manner, but also for lifting while in the rolled-on-its-side attitude for this test work. The test wing, ground plane, and floor-mounting adapter assembly would be mounted in the tunnel first; next, the nacelle less rotor blades would be mounted on the wing.

Since the nacelle consists of a fixed portion and a tilting/swiveling portion, and these must be shop-assembled prior to entering the test section; a temporary means is required to keep the two portions in relative fixed orientation during nacelle-wing assembly prior to the hook-up at the swivel quadrant assembly on the wing tip. A temporary tilt/swivel lock can be achieved by a boltup tab arrangement between the fixed nacelle structural adapter and the inboard and outboard trunnion mount fittings on the tilting rotor nacelle. The final items to be assembled would be the rotor blades and spinner. The two-pin retention design, between blade root and pitch shaft, outboard of the hub, should facilitate this final assembly.

A review will be made of the structural adequacy of the BHC 300-018-015-1 torque box and the adapter for use with the Boeing Vertol HTR XV-15 nacelle under loads pertinent to the test program.

2.6.2 Rotor

The rotor for use on the test stand will be a standard flight article except a special adapter will be provided on the hub to allow adding lubrication oil without removing the rotor from the test stand. See Appendix II.

2.6.3 Powered Test Stand Drive System

The powered nacelle test forces all the propulsion and drive train components (SK 27257, Figures 2.38 and 2.39), to lie on their left side. The drive train components are attitude sensitive only in relation to the lubrication system; in particular the drainage, scavenge and breather location are affected. Therefore, in the powered nacelle test special provisions will be made to assure satisfactory lubrication. In some instances, modification of the aircraft hardware will be made, in others, special test hardware will be substituted as explained below.

The engine box will be drained from two points. A large drain line will be connected to the inboard face of the housing, and a smaller line will be connected to the seal cap to drain this cavity. Provisions for both connections can be made in the hardware initially. In the righthand nacelle to be tested, the bevel gear is at the top of the box. This materially improves the scavenge situation by reducing churning and windage at the oil surface.

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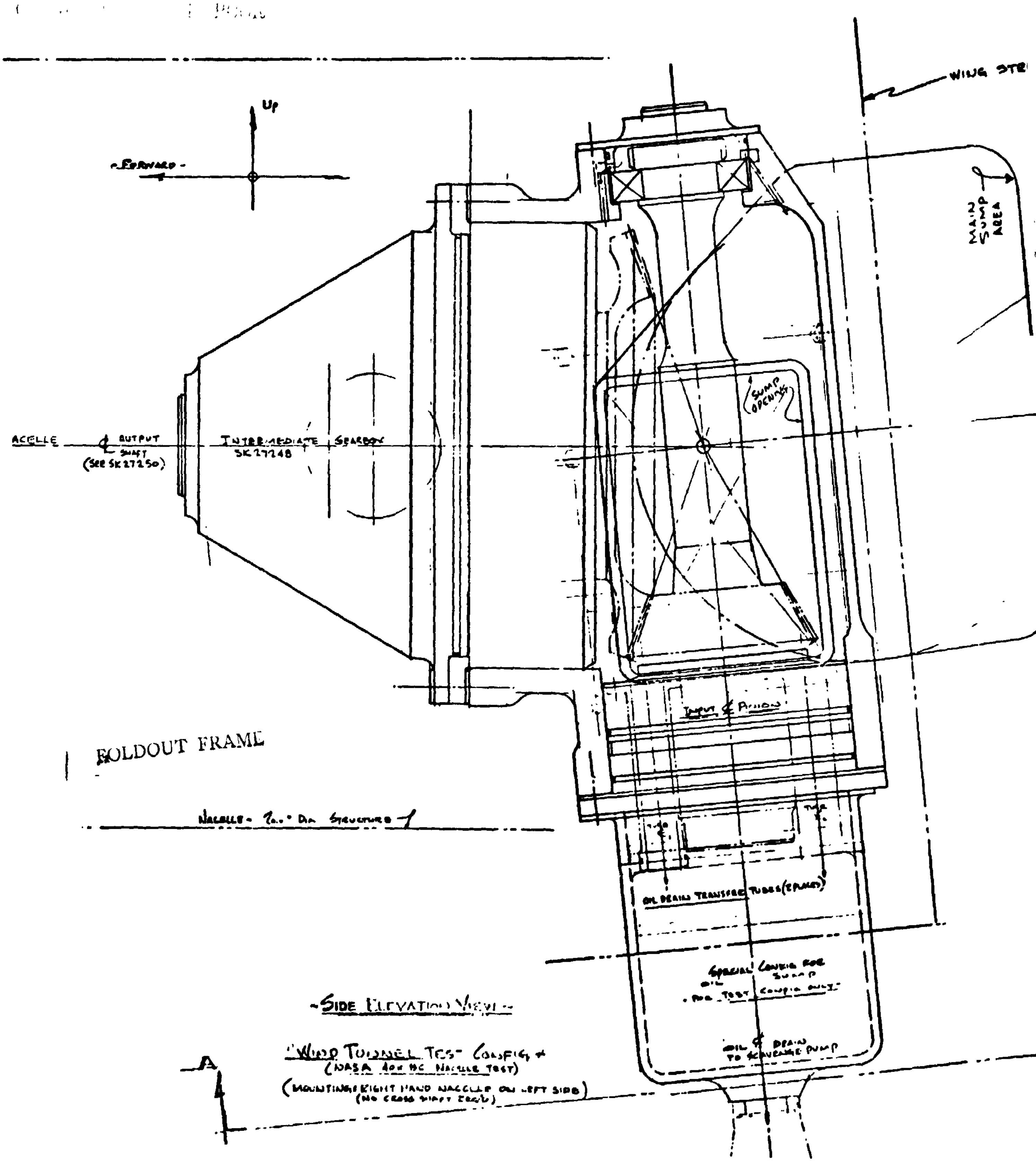

FOLDOUT FRAME

-SIDE ELEVATION VIEW-

"WIND TUNNEL TEST" CONFIG.

(NASA 40x80 NACELLE TEST)

(MOUNTING RIGHT HAND NACELLE ON LEFT SIDE)
(NO CROSS SHAFT REQ'D)

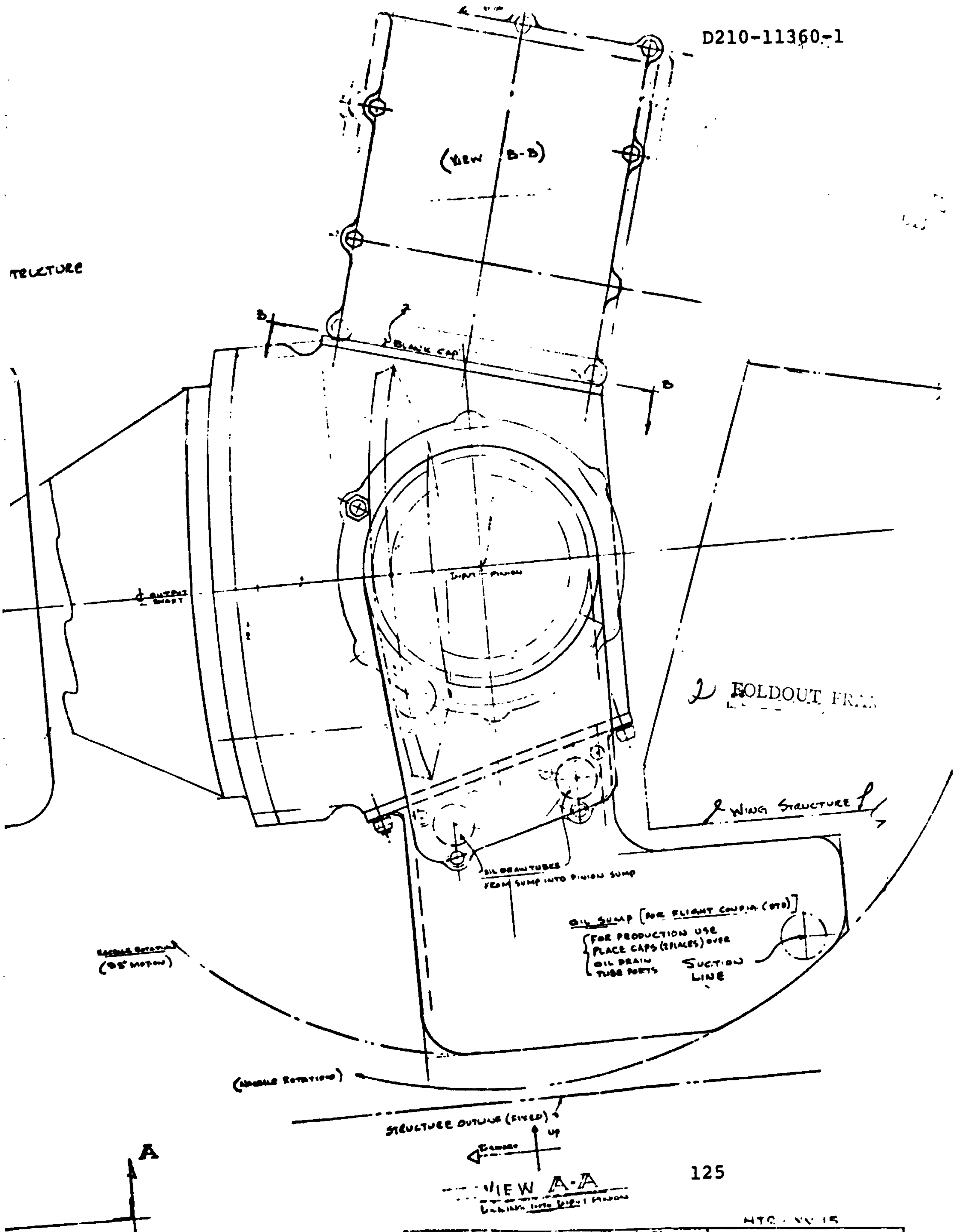

FIGURE 2.38. INTERMEDIATE GEARBOX TEST STAND CONFIGURATION - (NASA 40- X 80-FOOT TUNNEL)

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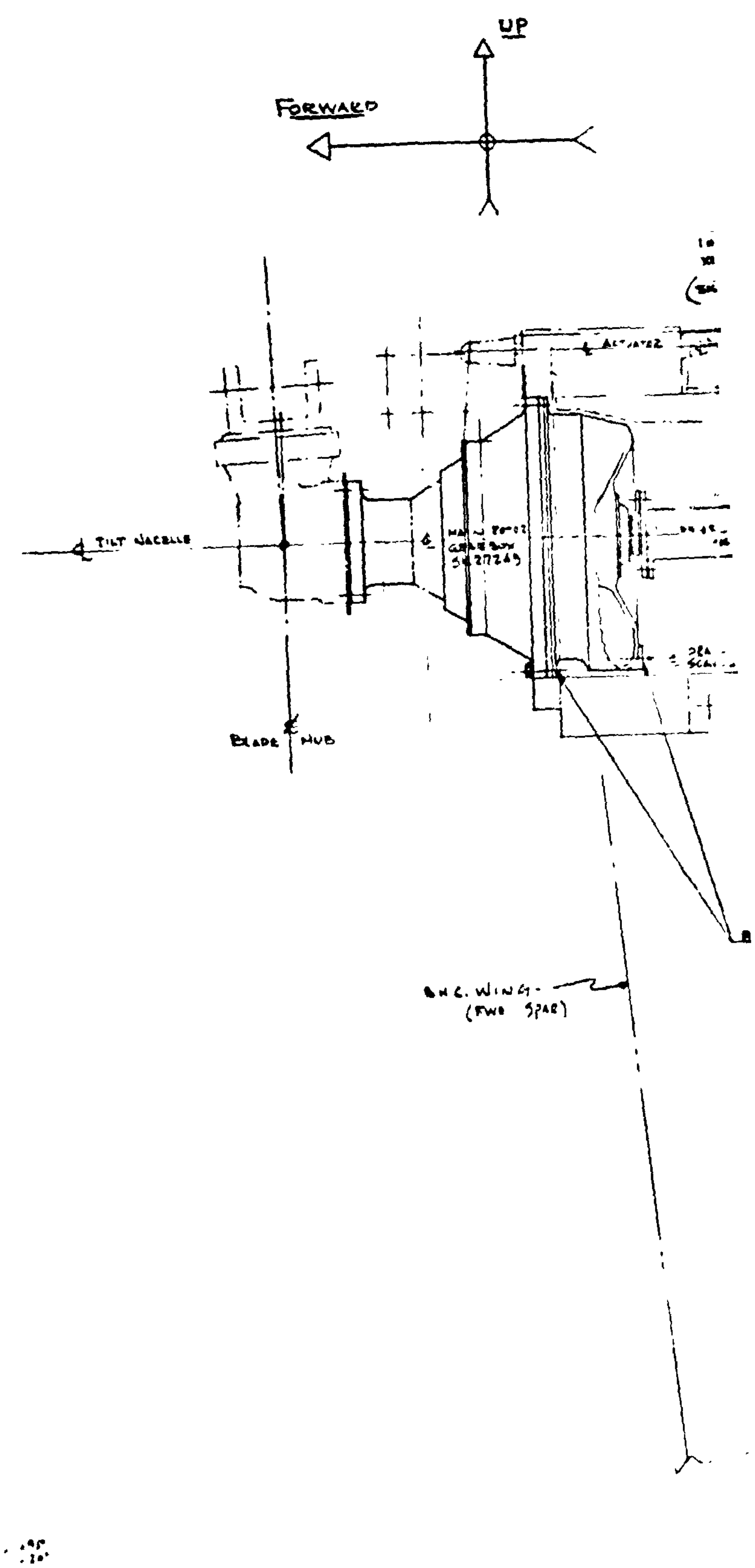


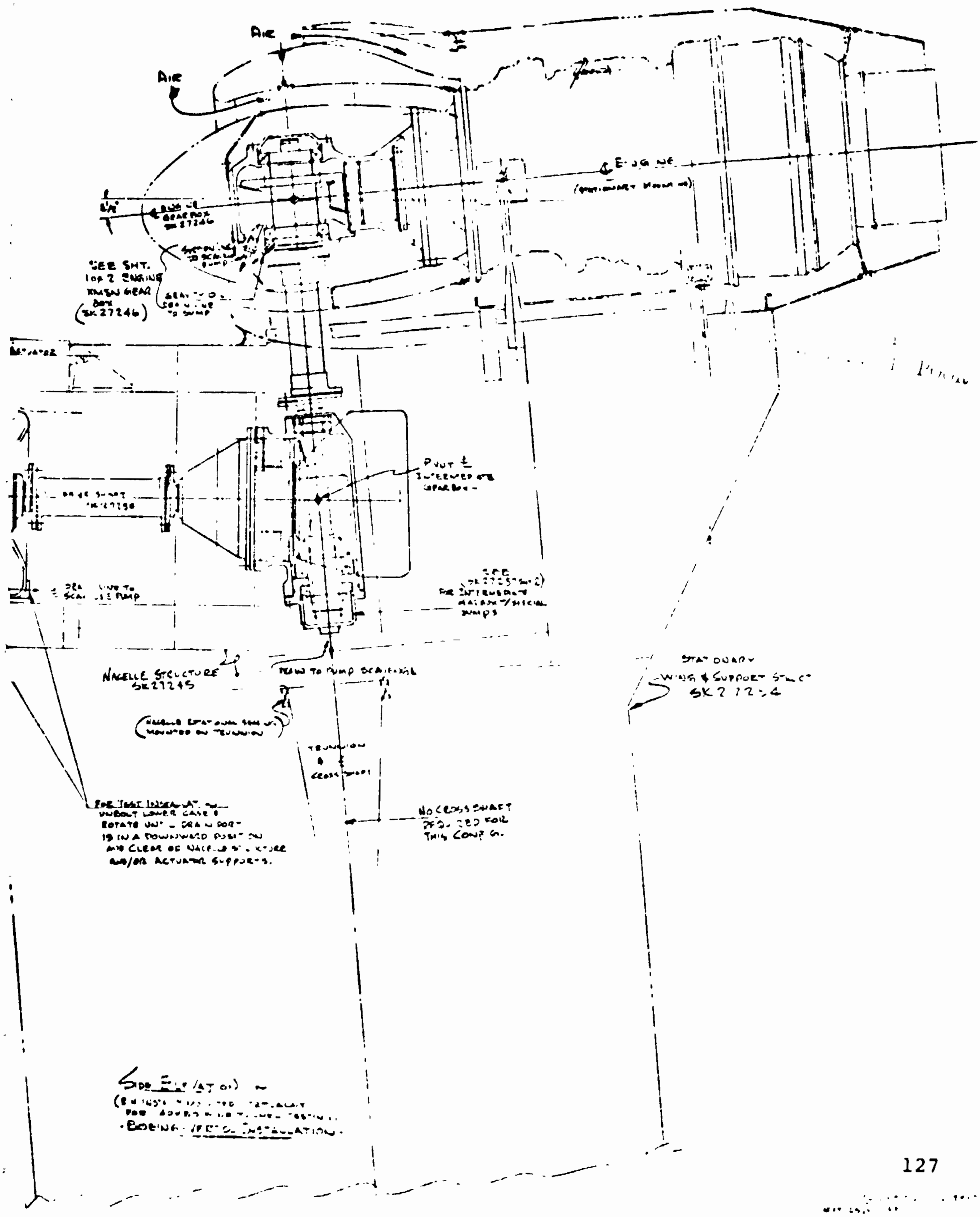


FIGURE 2.39. INSTALLATION - DRIVE SYSTEM - RIGHTHAND NACELLE MOUNTED VERTICALLY (XV-15 MODEL IN 40- X 80-FOOT TUNNEL)

The intermediate box will be drained from a special closed end cap covering the lower (inboard) shaft extension. Since no cross-shaft is used in the nacelle test, there is no requirement for a dynamic seal at this location. Additional drainage will be provided at the appropriate place in the aircraft sump.

The rotor box back cover will be rotated approximately 90° to locate the oil pickup point near the bottom at the test attitude. This will be accomplished with aircraft hardware.

Extensions will be added to gearbox breathers to raise them above oil level.

The accessory drive box will not be present in the nacelle test arrangement. Lubrication oil leaving the boxes will be routed down the wing to tanks located under the test floor. Oil supply to the boxes will be provided by facility pumps located near the tankage. Oil cooling will be provided by tube type coolers located near pumps. Filtration will also be provided.

In the absence of the accessory drive box, hydraulic and electrical power required to operate the test configuration will be supplied by the facility.

2.6.4 Control System for Wind Tunnel

For the Phase II Wind Tunnel Test the aircraft Primary Flight Control System (PFCS) will allow manual control of the three (3) rotor control actuators to allow setting of longitudinal and lateral cyclic, and collective pitch; and the control of one engine.

- The manual control of the rotor will be via redundant LVDTs identical to type planned for aircraft use. A suitable manual input device to position and lock the LVDTs will be produced. The PFCS panel and Maintenance Unit will be identical to aircraft configuration. The aircraft engine control quadrant will be used.
- The DEL Control Unit will be identical of the final aircraft design except that only the circuit cards and wiring necessary to control the single rotor will be installed.
- Rotor control actuators and their installation will be identical to the aircraft arrangement.
- Power will be ground based with adequate redundancy derived from batteries and hydraulic accumulators.

2.6.5 Powerplant for Wind Tunnel Test

The powerplant portion of the HTR XV-15 nacelles for test stand operation will be generally similar to the flight arrangement shown in Drawing SK-27244; however, certain changes are required. Since the engine itself requires changes, and flight engines are not likely to be available, another engine probably needs to be added to the program for this test stand work.

Engine - The LTC1K-4K engine has a normal roll attitude limit of 20°. Operation on a test stand rolled 90° on its side is feasible if the engine lubrication system is modified in the area of oil scavenging and oil system external elements. Discussions with Lycoming have indicated that from the engine viewpoint, it is best to mount the unit rolled onto its left side (looking forward) to most easily match engine and test stand-mounted lubrication components since engine scavenge pumps and lines are located on the left side. This dictates that testing be done on a righthand side nacelle (as regards installation on the aircraft). The normal engine oil tank in the nacelle lower lip would not be used. Oil drain and pressure lines would run aft of the rear spar down through the wing to an external system with a tank and an added boost pump. This means that revisions would have to be made in the engine compartment to an external system. The required holes would be later blocked in a revision to a flight system.

Mounting - The engine mounting system will remain unchanged from the flight system.

Cowling - No change from the flight system is envisioned except that drain lines will have to be revised for the on-the-side orientation. Since the cowl panels are arranged for engine removal vertically upward on the actual aircraft, the engine cannot be removed upward while on the test stand. Means must either be devised to remove the engine sideways or remove the whole nacelle.

Firewalls - Additional holes are required for the engine lubrication system lines, but the bleed air line can be omitted. It may be possible to pass the oil lines through the hole for the bleed air line. Bleed air is used to feed the aircraft ECU but is not required on the test stand.

Engine Air Induction - The engine inlet system is the same as the flight system.

Engine Air Exhaust - This is identical to the flight system.

Cooling - Both engine and engine compartment cooling on the test rig are like the flight system.

Starting - The 300 ampere DC engine starter/generators of the flight system should be satisfactory since DC electric starting power up to five to six times that amperage is available as a tunnel utility item. Electrical cabling would run to the starter on the engine from the tunnel power source up the section of the test wing aft of the torque box, and through the normal hole in the firewall for a starting cable from the flight aircraft battery system.

Lubrication - As noted above in Section 2.6 the engine lubrication system external to the engine is revised because of engine attitude on the test stand. The normal flight oil tank in the nacelle will not be used; an external (to the test stand) reservoir and pump will be added. Oil cooling via the fuel oil heat exchanger used on the flight aircraft should be satisfactory; if not, an external oil cooler can be added outside the test stand. Oil lines will run up and down the wing trailing edge section.

Fuel System - On the engine compartment side of the firewall, the fuel system of the test stand nacelle will be like the flight system, including filter and firewall shutoff valve. On the other side of the firewall, the fuel feed line will run down inside the wing aft of the torque box section, and will connect to the 40-foot by 80-foot wind tunnel fuel system supplying JP-5. The engine is compatible with this fuel type and the engine fuel inlet pressure and fuel flow requirements are well within the NASA Ames JP-5 fuel system maximum supply values of 50 gpm at 100 psi.

Fire Detection System - The system is the same as on the flight aircraft and will be connected to the appropriate tunnel indicators.

Fire Extinguishing System - This is identical to the flight nacelle system and will be connected to appropriate actuation controls in the tunnel.

Ice Protection System - No hookup of the engine inlet anti-icing system is anticipated for test work in the tunnel.

Engine Air Bleed - As noted in Section   above, air bleed is not required on the test stand, since its only flight function is to feed the flight aircraft environmental control unit.

2.6.6 Utility Subsystems

Hydraulic System - Hydraulic services in the test rig nacelle are required only for the rotor control power actuators - an element of the flight control system.

Electrical System - Electrical services to the nacelle are associated primarily with the engine and flight controls. Other electrical loads in the test nacelle are shown on Schematic Drawing SK-27256. These include the following, which will be serviced by the same electrical power supply.

a. Drive system transmission condition sensors.
b. Drive system torquemeter (for rotor drive only).
c. Nacelle lights.
d. Rotor telemetering transmitter.

Pneumatic - No pneumatic services are required.

2.6.7 Instrumentation

Normal Flight - This category of instrumentation covers those items normally included in the flight aircraft (normal instrument panel readout) separate and apart from test instrumentation normally required for research, which would also be included in the test stand arrangement. These instrumentation items will be hooked up to appropriate monitoring displays in the tunnel control room. Included are systems for displaying:

a. Transmission condition (temperature, pressure).

b. Engine condition (temperature, pressure).

c. Engine and rotor/drive tachometers.

d. Engine fuel flow.

e. Electrical system amperage and voltage.

f. Hydraulic system oil temperature and pressure.

g. Caution panel displays.

h. Torquemeter output.

i. Flight control system parameters.

Research Test Stand Instrumentation - This section identifies the parameters and transducers to be measured during the Phase II wind tunnel test other than the normal flight instrumentation listed above and detailed fly-by-wire control system measurements to be identified separately.

Rotating System -

a. Rotor - strain gage bending bridges for flap and chord bending at 12.5% radius on each blade to measure steady and alternating bending loads.

- blade angle transducer at blade root on one blade (i.e., rotary pot and rack and pinion drive).

b. Pitchlinks - strain gage bridges to measure steady and alternating pitchlink compression loads.

c. Shaft Torque - strain gages on rotor shaft.

d. Rotor Telemetry - Signals will be transmitted from the rotating to the fixed system, using an Acurex-Autodata Wireless Measurement System to provide a repatchable, four-channel system from each rotor.

- Acurex-Autodata Wireless Measurement System, Static Strain, Bandpass DC to 1 KHz, Four-Channel Capability, consisting of the following:

1. Model 155S-4 four-channel receiver. One each required.
2. Model 106S static strain signal conditioning cards. Four each required.
3. Model 206A static strain transmitter. Four each required.
4. Model 230A induced power regulator module. One each required.

5. Model 234A custom induced power matching coil. One each required.
6. Model 160A induction power oscillator. One each required.
7. Model 160PS power supply. One each required.
8. Model 1211S custom antenna matching network. One each required.

Fixed System

a. Actuators - rotor system actuator forces will be recorded using either strain gaged lugs or calibrating delta p across the piston and recording.

b. Nacelle Frame - six linear accelerometers distributed to measure the six components of linear and angular acceleration.

- strain gages at critical points.

c. Engine Mount - strain gages on all three mounting points.

- five accelerometers.

d. Wing - strain gage bridges as close to the tip as practical to measure chord and flap bending, torsion normal force (shear).

- similar set at the wing root.

e. Total System Needs - tunnel balance.

3.0 AIRCRAFT EVALUATION

This section of the report presents the results of the technology evaluation of the modified XV-15 aircraft described in Section 2.0 of this document. This includes the aircraft performance, flying qualities, the structural integrity of the airframe, weights, noise, and aeroelastic interactions. As might be expected, the critical technology questions in all of these areas are associated with the presence of the rotor. Of these, the question which is foremost is that of rotor-airframe dynamic stability at the high advance ratios which the tilt rotor must attain to achieve its unique cruise performance.

The predictability of rotor-airframe dynamics has been demonstrated in tests of dynamically similar models, and in tunnel tests of full scale flight worthy rotors operating at reduced tip speed to represent the higher cruise advance ratios. There is, therefore, a very high level of confidence that the demonstration of stability at maximum speed on an actual tilt rotor aircraft will be accomplished in the XV-15 flight test program. In each of the other technical areas there already exists a demonstrated methodology for predicting the behavior of the aircraft and its systems.

In the current work the technical evaluation of the HTR XV-15 has used these methods to provide the data given in the following paragraphs.

3.1 WEIGHTS

The mass properties data presented herein represents a summary of the weight and balance modifications associated with mounting a Boeing Vertol Hingeless Tilt Rotor (HTR) on the XV-15 V/STOL Tilt Rotor Research Aircraft. The delta weight empty associated with these modifications is +256.7 Kg (586 lb.). The XV-15 base weight empty used for the comparison was 4,116.9 Kg (9,076 lb). The HTR/XV-15 revised weight empty is 4,374 Kg (9,644 lb.).

Table 3.1.1 summarizes the HTR/XV-15 weight empty in MIL-STD-451 group weight statement format. Table 3.1.2 compares the group weights of the XV-15 and the HTR/XV-15 and identifies the weight differences. Additional details on the weight empty changes are included in Tables 3.1.3 through 3.1.6. Balance details for the fixed pylon and contents, engine controls and contents and the tilting pylon and contents are presented in Tables 3.1.7 through 3.1.9.

HTR/XV-15 design gross weight, balance and inertia data for mast angles between 0° and 90° at the XV-15 most forward and most aft hover CG limits are presented in Table 3.1.10.

A practical loading condition for the HTR/XV-15 6,154 Kg (13,568 lb.) design gross weight is shown in Table 3.1.11 with the critical points plotted on the existing XV-15 center of gravity limit diagram, Figure 3.1.1. These points are: (1) Weight empty 4374 Kg (9644 lb.), (2) Minimum flying weight 4988 Kg (10997 lb.), (3) with ballast to CG aft limit 5056 Kg (11,147 lb.), (4) with co-pilot 5147 Kg (11,307 lb.), (5) with maximum fuel 5741 Kg (12,657 lb.), (6) with payload 6154 Kg (13,568 lb.)

The HTR/XV-15 weights presented herein were determined by utilizing a combination of weight estimating techniques including theoretical stress analysis, weight trends, layout calculations, vendor quotations and the actual weights of existing aircraft components. Data is presented in both the international system of units (S.I.) and in U.S. units.

TABLE 3.1.1. GROUP WEIGHT STATEMENT - HTR/XV-15 - (S.I. UNITS)

GROUP	WEIGHT (Kg)		
Rotor Group			500.8
Blade Assembly		269.0	
Hub Assembly		201.0	
Spinner		30.8	
Wing Group			396.0
Tail Group			94.8
Horizontal Tail		55.3	
Vertical Tail		39.5	
Body Group			654.1
Alighting Gear			230.4
Flight Controls Group			476.3
Cockpit Controls		20.9	
Automatic Flight Controls System		47.2	
Rotor, Non-Rotating		75.3	
Rotor, Rotating		72.6	
Wing and Empennage		186.4	
Conversion System		73.9	
Engine Section			202.7
Fixed Structure		56.7	
Engine Nacelle		49.9	
Tilting Structure		96.1	
Propulsion Group			1,237.4
Engine Installation		492.2	
Air Induction		7.7	
Exhaust System		7.7	
Lubrication System		10.0	
Fuel System		87.1	
Engine Controls		18.6	
Starting System		43.5	
Drive System		570.6	
Gearboxes	488.1		
Interconnect Drive	40.3		
Rotor Drive	42.2		
Instrument Group			41.3
Hydraulic Group			117.9
Electrical Group			179.6
Furnishings and Equipment Group			175.1
Personnel Accommodations		127.0	
Miscellaneous Equipment & Furnishings		15.9	
Emergency Equipment		32.2	
Air Conditioning Equipment			45.4
Contingency			22.7
WEIGHT EMPTY - KILOGRAMS			4,374.5

TABLE 3.1.1.GROUP WEIGHT STATEMENT - HTR/XV-15 (U.S. UNITS)

GROUP	WEIGHTS		
Rotor Group			1,104
Blade Assembly		593	
Hub Assembly		443	
Spinner		68	
Wing Group			873
Tail Group			209
Horizontal Tail		122	
Vertical Tail		87	
Body Group			1,442
Alighting Gear			508
Flight Controls Group			1,050
Cockpit Controls		46	
Automatic Flight Controls System		104	
Rotor, Non-Rotating		166	
Rotor, Rotating		160	
Wing and Empennage		411	
Conversion System		163	
Engine Section			447
Fixed Structure		125	
Engine Nacelle		110	
Tilting Structure		212	
Propulsion Group			2,728
Engine Installation		1,089	
Air Induction		17	
Exhaust System		17	
Lubrication System		22	
Fuel System		192	
Engine Controls		41	
Starting System		96	
Drive System		1,258	
Gearboxes	1,076		
Interconnect Drive	89		
Rotor Drive	93		
Instrument Group			91
Hydraulic Group			260
Electrical Group			396
Furnishings and Equipment Group			386
Personnel Accommodations		280	
Miscellaneous Equipment & Furnishings		35	
Emergency Equipment		71	
Air Conditioning Equipment			100
Contingency			50
WEIGHT EMPTY - POUNDS			9,644

TABLE 3.1.2. GROUP WEIGHT STATEMENT COMPARISON
(S.I. UNITS)

GROUP	XV-15	DELTA WT (Kg)	HTR/XV-15
Rotor Group	406.9	(+93.9)	500.8
Blade Assembly	256.3	+12.7	269.0
Hub Assembly	123.4	+77.6	201.0
Spinner	27.2	+3.6	30.8
Wing Group	396.0		396.0
Tail Group	94.8		94.8
Horizontal Tail	55.3		55.3
Vertical Tail	39.5		39.5
Body Group	654.1		654.1
Alighting Gear	230.4		230.4
Flight Controls Group	458.1	(+18.2)	476.3
Cockpit Controls	20.9		20.9
Automatic Flight Controls System	80.1	-32.9	47.2
Rotor, Non-Rotating	137.9	-62.6	75.3
Rotor, Rotating	78.5	-5.9	72.6
Wing and Empennage	66.7	+119.[illegible]	186.4
Conversion System	74.0	-.1	73.9
Engine Section	128.8	(+73.9)	202.7
Engine Mount/Fixed Structure	-	+56.7	56.7
Firewall/Engine Nacelle	11.3	+38.6	49.9
Cowl/Tilting Structure	117.5	-21.4	96.1
Propulsion Group	1,165.8	(+71.6)	1,237.4
Engine Installation	492.2		492.2
Air Induction	7.7		7.7
Exhaust System	7.7		7.7
Lubrication System	10.0		10.0
Fuel System	87.1		87.1
Engine Controls	18.6		18.6
Starting System	43.5		43.5
Drive System	499.0	+71.6	570.6
Gearboxes	445.9		488.1
Interconnect Drive	23.6		40.3
Rotor Drive	[illegible].5		42.2
Instrument Group	41.3		41.3
Hydraulic Group	117.9		117.9
Electrical Group	179.6		179.6
Furnishings and Equipment Group	175.1		175.1
Personnel Accommodations	127.0		127.0
Miscellaneous Equipment and Furnishings	15.9		15.9
Emergency Equipment	32.2		32.2
Air Conditioning Equipment	45.4		45.4
Contingency	22.7		22.7
WEIGHT EMPTY - KILOGRAMS	4,116.9	+257.6	4,374.5

TABLE 3.1.2. GROUP WEIGHT STATEMENT COMPARISON
(U.S. UNITS)

GROUP	XV-15		DELTA WEIGHT	HTR/XV-15	
Rotor Group		897	(+207)		1,104
Blade Assembly	565		+ 28	593	
Hub Assembly	272		+171	443	
Spinner	60		+ 8	68	
Wing Group		873			873
Tail Group		209			209
Horizontal Tail	122			122	
Vertical Tail	87			87	
Body Group		1,442			1,442
Alighting Gear		508			508
Flight Controls Group		1,010	(+ 40)		1,050
Cockpit Controls	46			46	
Automatic Flight Controls System	177		- 73	104	
Rotor, Non-Rotating	304		-138	166	
Rotor, Rotating	173		- 13	160	
Wing and Empennage	147		+264	411	
Conversion System	163			163	
Engine Section		284	(+163)		447
Engine Mount/Fixed Structure	-		+125	125	
Firewall/Engine Nacelle	25		+ 85	110	
Cowl/Tilting Structure	259		- 47	212	
Propulsion Group		2,570	(+158)		2,728
Engine Installation	1,085			1,085	
Air Induction	17			17	
Exhaust System	17			17	
Lubrication System	22			22	
Fuel System	192			192	
Engine Controls	41			41	
Starting System	96			96	
Drive System	1,100		+158	1,258	
Gearboxes	983			1,076	
Interconnect Drive	52			89	
Rotor Drive	65			93	
Instrument Group		91			91
Hydraulic Group		260			260
Electrical Group		396			396
Furnishings and Equipment Group		386			386
Personnel Accommodations	280			280	
Miscellaneous Equipment and Furnishings	35			35	
Emergency Equipment	71			71	
Air Conditioning Equipment		100			100
Contingency		50			50
WEIGHT EMPTY - POUNDS		9,076	+568		9,644

TABLE 3.1.3. ROTOR GROUP WEIGHT SUMMARY

(S.I. UNITS)

GROUP	WEIGHT (Kg)	
ROTOR GROUP		500.8
. BLADES (6) (INCLUDES 1.9 Kg TUNING WEIGHT, AND 1.8 Kg TIP WEIGHT PER BLADE)		269.0
. HUB AND RETENTION (2)		200.9
ROTOR HUB (2)	58.1	
PITCH SHAFT (6)	53.4	
CENTER BLOCK (2)	3.6	
RETENTION POST (6)	13.5	
RETENTION PIN (6)	2.3	
ELASTOMERIC BEARING (6)	19.5	
BLADE ATTACHMENT PIN (12)	11.8	
PIN BOLT (12)	.5	
PIN CAP (12)	.5	
OUTBOARD BEARING (6)	10.8	
INBOARD BEARING (6)	7.7	
OUTBOARD LINER (6)	3.6	
INBOARD LINER (6)	1.4	
MOUNT BUSHING (24)	.9	
MOUNT STUD (24)	2.7	
MOUNT SPACER (2)	.9	
BEARING RETAINER (6)	1.4	
LOWER POSITIONER (2)	.5	
RESERVOIR (2)	.5	
OUTBOARD SEAL AND RACE (6)	1.3	
INBOARD SEAL (6)	.5	
HARDWARE, ETC.	2.3	
OIL	3.2	
. SPINNERS (2)		30.9

TABLE 3.1.3. ROTOR GROUP WEIGHT SUMMARY

(U.S. UNITS)

GROUP	WEIGHT (LB)		
ROTOR GROUP			1,104
. BLADES (6) (INCLUDES 4.2 POUNDS TUNING WEIGHT, AND 4.0 POUNDS TIP WEIGHT PER BLADE)		593	
. HUB AND RETENTION (2)		443	
ROTOR HUB (2)	128		
PITCH SHAFT (6)	118		
CENTER BLOCK (2)	8		
RETENTION POST (6)	30		
RETENTION PIN (6)	5		
ELASTOMERIC BEARING (6)	43		
BLADE ATTACHMENT PIN (12)	26		
PIN BOLT (12)	1		
PIN CAP (12)	1		
OUTBOARD BEARING (6)	24		
INBOARD BEARING (6)	17		
OUTBOARD LINER (6)	8		
INBOARD LINER (6)	3		
MOUNT BUSHING (24)	2		
MOUNT STUD (24)	6		
MOUNT SPACER (2)	2		
BEARING RETAINER (6)	3		
LOWER POSITIONER (2)	1		
RESERVOIR (2)	1		
OUTBOARD SEAL AND RACE (6)	3		
INBOARD SEAL (6)	1		
HARDWARE, ETC.	5		
OIL	7		
. SPINNERS (2)		68	

TABLE 3.1.4. FLIGHT CONTROLS WEIGHT SUMMARY (S.I. UNITS)

-FLY-BY-WIRE SYSTEM-

GROUP	WEIGHT (Kg)		
FLIGHT CONTROLS GROUP			476.3
COCKPIT CONTROLS		20.9	
AUTOMATIC FLIGHT CONTROLS		47.2	
. CONTROL UNITS (2)	10.9		
. FORCE FEEL ACTUATORS	27.2		
. CENTERING SPRINGS	3.2		
. MAGNETIC BRAKES	4.5		
. VISCOUS DAMPERS	1.4		
NON-ROTATING - ROTOR		75.3	
. UPPER CONTROL ACTUATORS (6)	68.0		
. SCISSORS	7.3		
ROTATING-ROTOR		72.6	
. SWASHPLATE	38.6		
. SCISSORS	4.1		
. PITCH LINKS	16.8		
. SUPPORTS AND MISC	13.1		
WING AND EMPENNAGE		186.4	
. WING & TAIL ACTUATORS (4)	45.3		
. CONTROL ELECTRONICS	45.3		
. THIRD CHANNEL GENERATOR	6.8		
. GENERATOR CONTROL	1.4		
. JUNCTION BOXES (3)	5.4		
. ROTOR RPM TRANSDUCERS (6)	.5		
. NACELLE ANGLE TRANSDUCERS (2)	1.4		
. LVDT TRANSDUCERS	5		
. WIRING	[illegible]		
. PANELS	[illegible]		
. SUPPORTS AND MISCELLANEOUS	22.7		
CONVERSION SYSTEM			
. SPINDLE INSTALLATION, ACTUATORS, ETC.		73.9	

TABLE 3.1.4.FLIGHT CONTROLS WEIGHT SUMMARY

-FLY-BY-WIRE SYSTEM-
(U.S. UNITS)

GROUP	WEIGHT (LB)		
FLIGHT CONTROLS GROUP			1,050
COCKPIT CONTROLS		46	
AUTOMATIC FLIGHT CONTROLS		104	
. CONTROL UNITS (2)	24		
. FORCE FEEL ACTUATORS	60		
. CENTERING SPRINGS	7		
. MAGNETIC BRAKES	10		
. VISCOUS DAMPERS	3		
NON-ROTATING - ROTOR		166	
. UPPER CONTROL ACTUATORS (6)	150		
. SCISSORS	16		
ROTATING-ROTOR		160	
. SWASHPLATE	85		
. SCISSORS	9		
. PITCH LINKS	37		
. SUPPORTS AND MISC	29		
WING AND EMPENNAGE		411	
. WING & TAIL ACTUATORS (4)	100		
. CONTROL ELECTRONICS	100		
. THIRD CHANNEL GENERATOR	15		
. GENERATOR CONTROL	3		
. JUNCTION BOXES (3)	12		
. ROTOR RPM TRANSDUCERS (6)	1		
. NACELLE ANGLE TRANSDUCERS (2)	3		
. LVDT TRANSDUCERS	10		
. WIRING	112		
. PANELS	5		
. SUPPORTS AND MISCELLANEOUS	50		
CONVERSION SYSTEM		163	
. SPINDLE INSTALLATION, ACTUATORS, ETC.			

TABLE 3.1.5. ENGINE SECTION WEIGHT SUMMARY
(S.I. UNITS)

GROUP	WEIGHT (Kg)		
ENGINE SECTION			202.8
FIXED STRUCTURE		56.7	
. MAIN BOX (2)	42.6		
. AFT BOX (2)	5.4		
. FAIRING & DOORS (2)	8.7		
ENGINE NACELLE (2)		49.9	
TILTING STRUCTURE		96.2	
. STRUCTURE (2)	75.3		
. FAIRINGS (2)	20.9		

TABLE 3.1.5. ENGINE SECTION WEIGHT SUMMARY

(U.S. UNITS)

GROUP	WEIGHT (LB)		
ENGINE SECTION			447
FIXED STRUCTURE		125	
. MAIN BOX (2)	94		
. AFT BOX (2)	12		
. FAIRING & DOORS (2)	19		
ENGINE NACELLE (2)		110	
TILTING STRUCTURE		212	
. STRUCTURE (2)	166		
. FAIRINGS (2)	46		

TABLE 3.1.6. DRIVE SYSTEM WEIGHT SUMMARY

(S.I. UNITS)

GROUP	WEIGHT (Kg)		
DRIVE SYSTEM			570.6
GEARBOXES		488.1	
MAIN	146.5		
INTERMEDIATE	118.8		
ENGINE	75.8		
CENTER	30.4		
ACCESSORY	22.7		
LUB	85.7		
FAIRING AND SUPPORTS	8.2		
INTERCONNECT DRIVE		40.4	
WING	23.6		
ENGINE TO INTERMEDIATE BOX	8.6		
INTERMEDIATE TO MAIN BOX	8.2		
ROTOR DRIVE		42.1	

TABLE 3.1.6. DRIVE SYSTEM WEIGHT SUMMARY
(U.S. UNITS)

GROUP	WEIGHT (LB)		
DRIVE SYSTEM			1,258
GEARBOXES		1,076	
MAIN	323		
INTERMEDIATE	262		
ENGINE	167		
CENTER	67		
ACCESSORY	50		
LUB	189		
FAIRING AND SUPPORTS	18		
INTERCONNECT DRIVE		89	
WING	52		
ENGINE TO INTERMEDIATE BOX	19		
INTERMEDIATE TO MAIN BOX	18		
ROTOR DRIVE		93	

TABLE 3.1.7 FIXED PYLON AND CONTENTS (INBOARD)

(S.I. UNITS)

GROUP	WEIGHT PER SIDE	FUSELAGE STATION (X)	BUTTLINE (Y)	WATER-LINE (Z)
	(Kg)	(M)	(M)	(M)
ENGINE SECTION-FIXED	(28.6)	(7.93)	(5.11)	(2.57)
. MAIN STRUCTURE	21.3	7.85	5.08	2.57
. AFT STRUCTURE	2.7	8.34	5.41	2.60
. FAIRING	4.6	8.06	5.05	2.55
MISC. FIXED EQUIPMENT	(22.2)	(8.72)	(5.05	(2.57)
TOTAL PER SIDE	50.8	8.28	5.08	2.52

(U.S. UNITS)

GROUP	WEIGHT PER SIDE	FUSELAGE STATION (X)	BUTTLINE (Y)	WATER-LINE (Z)
	(LB)	(IN.)	(IN.)	(IN.)
ENGINE SECTION-FIXED	(63)	(312.1)	(201.1)	(101.3)
. MAIN STRUCTURE	47	308.9	200	101.3
. AFT STRUCTURE	6	328.4	213	102.3
. FAIRING	10	317.4	199	100.3
MISC. FIXED EQUIPMENT	(49)	(343.4)	(199)	(101.3)
TOTAL PER SIDE	112	325.8	200.2	99.2

TABLE 3.1.8. ENGINE COWL AND CONTENTS (OUTBOARD)
(S.I. UNITS)

	WEIGHT PER SIDE	FUSELAGE STATION (X)	BUTTLINE (Y)	WATER-LINE (Z)
	(Kg)	(M)	(M)	(M)
ENGINE SECTION - FIXED	(24.9)	(8.32)	(5.83)	(2.55)
. COWLING	24.9	8.32	5.83	2.55
PROPULSION	(280.8)	(8.52)	(5.83)	(2.55)
. ENGINE	245.9	8.52	5.83	2.55
. AIR INDUCTION	4.1	7.83	5.79	2.55
. EXHAUST	4.1	9.13	5.93	2.55
. LUBRICATION	5.0	8.52	5.83	2.55
. STARTING SYSTEM	21.7	8.52	5.83	2.55
DRIVE SYSTEM	(46.3)	(7.53)	(5.76)	(2.55)
. ENGINE BOX	37.7	7.53	5.79	2.55
. DRIVE SYSTEM	4.5	7.58	5.46	2.55
. SUPPORTS & FAIRING	4.1	7.53	5.79	2.55
TOTAL PER SIDE	352	8.37	5.82	2.55

(U.S. UNITS)

	(LB)	(IN.)	(IN.)	(IN.)
ENGINE SECTION - FIXED	(55)	(327.4)	(229.5)	(100.3)
. COWLING	55	327.4	229.5	
PROPULSION	(619)	(335.4)	(229.5)	(100.3)
. ENGINE	542	335.4	229.5	100.3
. AIR INDUCTION	9	308.4	228.0	100.3
. EXHAUST	9	359.4	233.0	100.3
. LUBRICATION	11	335.4	229.5	100.3
. STARTING SYSTEM	48	335.4	229.5	100.3
DRIVE SYSTEM	(102)	(296.6)	(226.7)	(100.3)
. ENGINE BOX	83	296.4	228.0	100.3
. DRIVE SYSTEM	10	298.4	215.0	100.3
. SUPPORTS & FAIRING	9	296.4	228.0	100.3
TOTAL PER SIDE	776	329.7	229.2	100.3

TABLE 3.1.9. TILTING PYLON AND CONTENTS

(S.I. UNITS)

GROUP	WEIGHT PER SIDE (Kg)	FUSELAGE STATION (X) M	BUTT LINE (Y) M	WATERLINE (Z) M
ROTOR	250.4	(6.21)	(5.05)	(2.55)
. BLADES, HUB & RETENTION	260.5	6.21	5.05	2.55
FLIGHT CONTROLS	81.6	(6.73)	(5.05)	2.48
. SWASHPLATE	28.1	6.47	5.05	2.55
. PITCH LINKS	8.2	6.39	5.05	2.55
. ACTUATORS (3)	34.0	6.77	5.05	2.55
. HYDRAULIC PUMPS	11.3	7.50	5.05	2.04
ENGINE SECTION	48.1	(7.07)	(5.05)	(2.49)
. STRUCTURE	32.2	7.07	5.05	2.55
. FAIRINGS) . MISCELLANEOUS)	15.9	7.08	5.05	2.35
DRIVE SYSTEM	196.4	(7.00)	(5.05)	(2.42)
. MAIN BOX	56.7	6.64	5.05	2.55
. INTERMEDIATE BOX	59.4	7.50	5.05	2.55
. ACCESSORY BOX	11.3	7.35	5.05	2.07
. ROTOR SHAFT	21.3	6.46	5.05	2.55
. DRIVE SHAFT	4.1	7.00	5.05	2.55
. LUBRICATION	37.7	6.94	5.05	2.07
. MISCELLANEOUS	5.9	7.00	5.05	2.42
TOTAL PER SIDE (NACELLE HORIZONTAL)	576.5	6.62	5.05	2.49
TOTAL PER SIDE (NACELLE VERTICAL)	576.5	7.55	5.05	3.53

TABLE 3.1.9. TILTING PYLON AND CONTENTS

(U.S. UNITS)

GROUP	WEIGHT PER SIDE LB	FUSELAGE STATION (X) IN.	BUTT LINE (Y) IN.	WATERLINE (Z) IN.
ROTOR	(552)	(244.4)	(199)	(100.3)
. BLADES, HUB & RETENTION	552	244.4	199	100.3
FLIGHT CONTROLS	(180)	(264.9)	(199)	(97.5)
. SWASHPLATE	62	254.6	199	100.3
. PITCH LINKS	18	251.4	199	100.3
. ACTUATORS (3)	75	266.4	199	100.3
. HYDRAULIC PUMPS	25	295.4	199	80.3
ENGINE SECTION	(106)	(278.3)	(199)	(97.9)
. STRUCTURE	71	278.4	199	100.3
. FAIRINGS) . MISCELLANEOUS)	35	278.9	199	92.5
DRIVE SYSTEM	(433)	(275.6)	199	(95.4)
. MAIN BOX	125	261.4	199	100.3
. INTERMEDIATE BOX	131	295.4	199	100.3
. ACCESSORY BOX	25	289.4	199	81.3
. ROTOR SHAFT	47	254.4	199	100.3
. DRIVE SHAFT	9	275.4	199	100.3
. LUBRICATION	83	273.4	199	81.3
. MISCELLANEOUS	13	275.6	199	95.4
TOTAL PER SIDE (NACELLE HORIZONTAL)	1,271	260.8	199	98.2
TOTAL PER SIDE (NACELLE VERTICAL)	1,271	297.3	199	138.9

TABLE 3.1.10. WEIGHT AND INERTIA DATA
(S.I. UNITS)

Weight (Kg)	Mast Angle Degs.	Fuselage Station (X) M	Water-Line (Z) M	I_X (Roll) (Kg M^2)	I_Y (Pitch) (Kg M^2)	I_Z (Yaw) (Kg M^2)
AFT C.G. (TOTAL AIRCRAFT)						
	0	7.57	1.95	70,956	19,659	83,848
	30	7.56	2.05	71,623	20,138	83,843
6,154	60	7.59	2.15	72,590	20,560	83,825
	75	7.61	2.19	75,003	20,708	83,817
	90	7.65	2.22	73,258	20,799	83,811
FORWARD C.G. (TOTAL AIRCRAFT)						
	0	7.33	1.95	70,956	19,609	84,922
	30	7.32	2.05	71,623	20,066	84,737
6,154	60	7.34	2.15	72,590	20,563	84,245
	75	7.38	2.19	75,004	20,885	84,051
	90	7.41	2.22	73,258	20,963	83,976

PYLON DATA

	Mast Angle - Degrees 0°	90°
WEIGHT PER SIDE (Kg)	576.5	
CENTER OF GRAVITY F.S. (X)	6.62	7.55
B.L. (Y)	5.05	5.05
W.L. (Z)	2.49	3.52
INERTIAS (Kg M^2)		
I_{XX} (ROLL)	.47	2.49
I_{YY} (PITCH)	2.74	2.74
I_{ZZ} (YAW)	2.49	.47

HORIZONTAL (X) CONVERSION MOMENT FROM 0° TO 90° = 1,073 Kg M (PER AIRCRAFT)

TABLE 3.1.10. WEIGHT AND INERTIA DATA
(U.S. UNITS)

Weight Lbs.	Mast Angle Degs.	Fuselage Station (X) IN.	Water-Line (Z) IN.	I_X (Roll) Slug-Ft^2	I_Y (Pitch) Slug-Ft^2	I_Z (Yaw) Slug-Ft^2
AFT C.G. (TOTAL AIRCRAFT)						
	0	298.2	76.7	52,293	14,488	61,794
	30	297.7	80.7	52,784	14,841	61,790
13,568	60	298.8	84.7	53,497	15,152	61,777
	75	299.8	86.2	53,802	15,261	61,771
	90	301.2	87.3	53,989	15,328	61,767
FORWARD C.G. (TOTAL AIRCRAFT)						
	0	288.7	76.7	52,293	14,451	62,585
	30	288.2	80.7	52,784	14,788	62,449
13,568	60	289.3	84.7	53,497	15,154	62,086
	75	290.4	86.2	53,802	15,392	61,943
	90	291.7	87.3	53,989	15,449	61,888

PYLON DATA

	Mast Angle - Degrees 0°	90°
WEIGHT PER SIDE (LBS)	1,271	
CENTER OF GRAVITY F.S. (X)	260.8	297.3
B.L. (Y)	199.0	199.0
W.L. (Z)	98.2	138.9
INERTIAS (SLUG-FT^2)		
I_{XX} (ROLL)	18.7	98.0
I_{YY} (PITCH)	108.0	108.0
I_{ZZ} (YAW)	98.0	18.7

HORIZONTAL (X) CONVERSION MOMENT FROM 0° TO 90° = 92,783 LBS. IN. (PER AIRCRAFT)

TABLE 3.1.11. DESIGN GROSS WEIGHT AND BALANCE SUMMARY

(S.I. UNITS)

GRAPH POINT	GROUP	WEIGHT (Kg)	STATION (X) M	MOMENT (Kg M)
1	WEIGHT EMPTY	4,374.5	(7.79)	34,077
	. PILOT (1)	90.7	5.28	479
	. TRAPPED LIQUIDS & OIL			
	ENGINE OIL	24.1	7.76	187
	TRAPPED ENGINE OIL	1.8	8.38	15
	UNUSABLE FUEL	8.6	7.73	66
	. OXYGEN	27.2	4.18	113
	. RESEARCH INSTRUMENTATION	339.7	7.85	2,666
	FIXED	136.1		
	PORTABLE	203.6		
	. AVIONICS AND NAVIGATION	65.4	6.17	403
	. ENVIRONMENTAL CONTROLS PACKAGE	12.2	10.87	132
	. CONTROL SHAKER INSTALL-ATION	7.7	8.20	63
	. FUEL	36.3	7.73	280
2	MINIMUM FLYING WEIGHT	4,988.2	(7.71)	38,481
	. BALLAST	68.0	3.45	234
3		5,056.2	(7.66)	38,715
	. CO-PILOT	90.7	5.28	478
4		5,146.9	(7.62)	39,193
	. FUEL (MAX)	594.3	7.52	4,469
5		5,741.2	(7.61)	43,662
	. PAYLOAD	413.2	8.05	3,326
6	DESIGN GROSS WEIGHT	6,154.4	(7.63)	46,988

TABLE 3.1.11. DESIGN GROSS WEIGHT AND BALANCE SUMMARY

(U.S. UNITS)

GRAPH POINT	ITEMS	WEIGHT (LB)	STATION (INCHES)	MOMENT (IN.-LB)
1	WEIGHT EMPTY	9,644	306.8	2,958,375
	. PILOT (1)	200	208.0	41,600
	. TRAPPED LIQUIDS & OIL			
	ENGINE OIL	53	305.4	16,186
	TRAPPED ENGINE OIL	4	330.0	1,320
	UNUSABLE FUEL	19	304.2	5,780
	. OXYGEN	60	164.6	9,875
	. RESEARCH INSTRUMENTATION	749	309.3	231,650
	FIXED	300		
	PORTABLE	449		
	. AVIONICS AND NAVIGATION	144	243.0	34,994
	. ENVIRONMENTAL CONTROLS PACKAGE	27	428.0	11,428
	. CONTROL SHAKER INSTALLATION	17	323.0	5,465
	. FUEL	80	304.2	24,336
2	MINIMUM FLYING WEIGHT	10,997	(303.8)	3,341,009
	. BALLAST	150	136	20,400
3		11,147	(301.5)	3,361,409
	. CO-PILOT	200	208.0	41,600
4		11,347	(299.9)	3,403,009
	. FUEL (MAX)	1,310	296.3	388,077
5		12,657	(299.5)	3,791,086
	. PAYLOAD	911	317.0	288,787
6	DESIGN GROSS WEIGHT	13,568	(300.7)	4,079,873

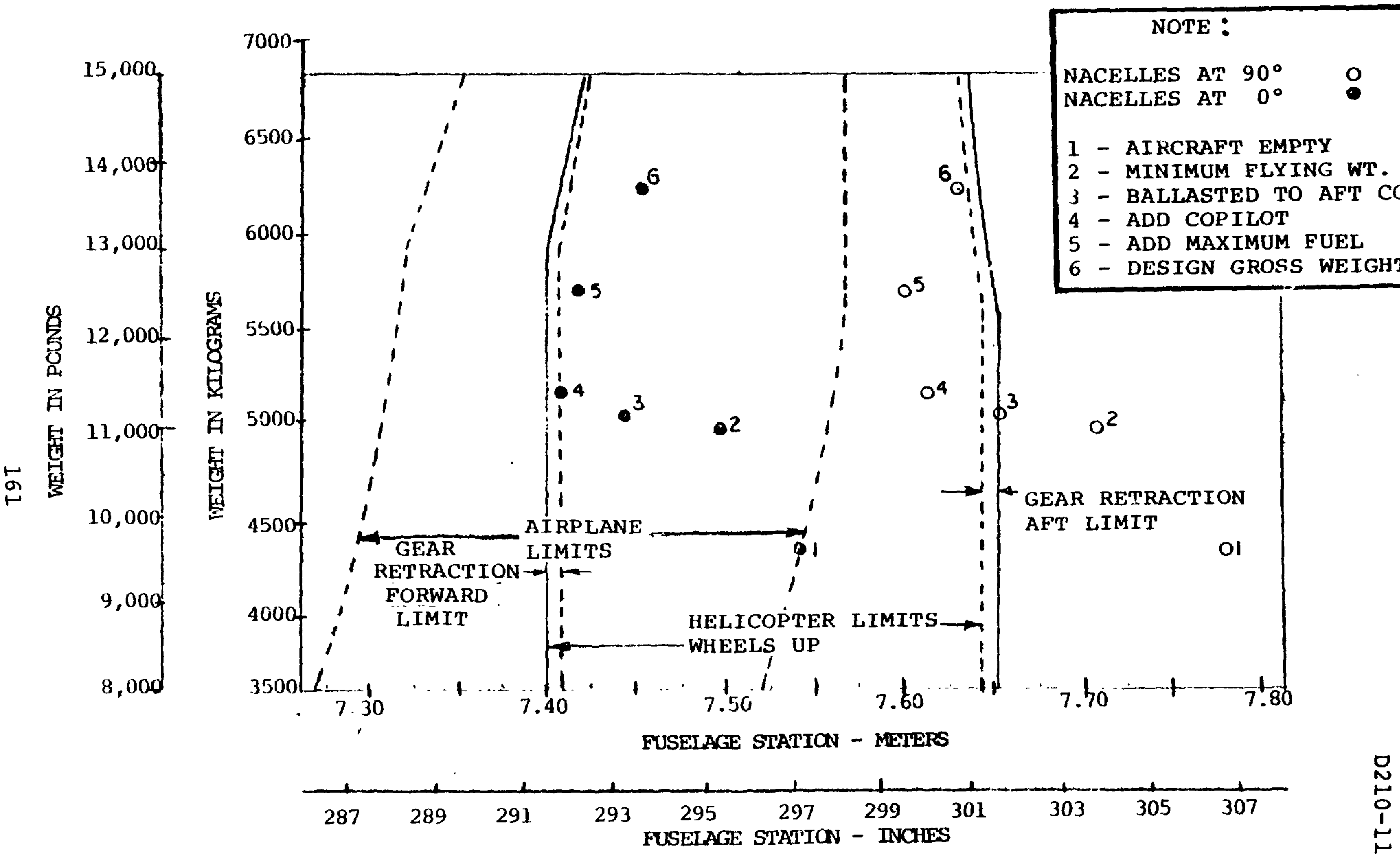


FIGURE 3.1.1. HTR XV-15 CENTER OF GRAVITY LIMITS AND SELECTED LOADING CONDITIONS

3.2 PERFORMANCE

This subsection presents the estimated performance for the HTR XV-15. The performance was computed following a review and updating of the following areas to reflect the fixed-engine configuration.

- o Engine Performance
- o Airframe Aerodynamics
- o Weights

3.2.1 Engine Performance

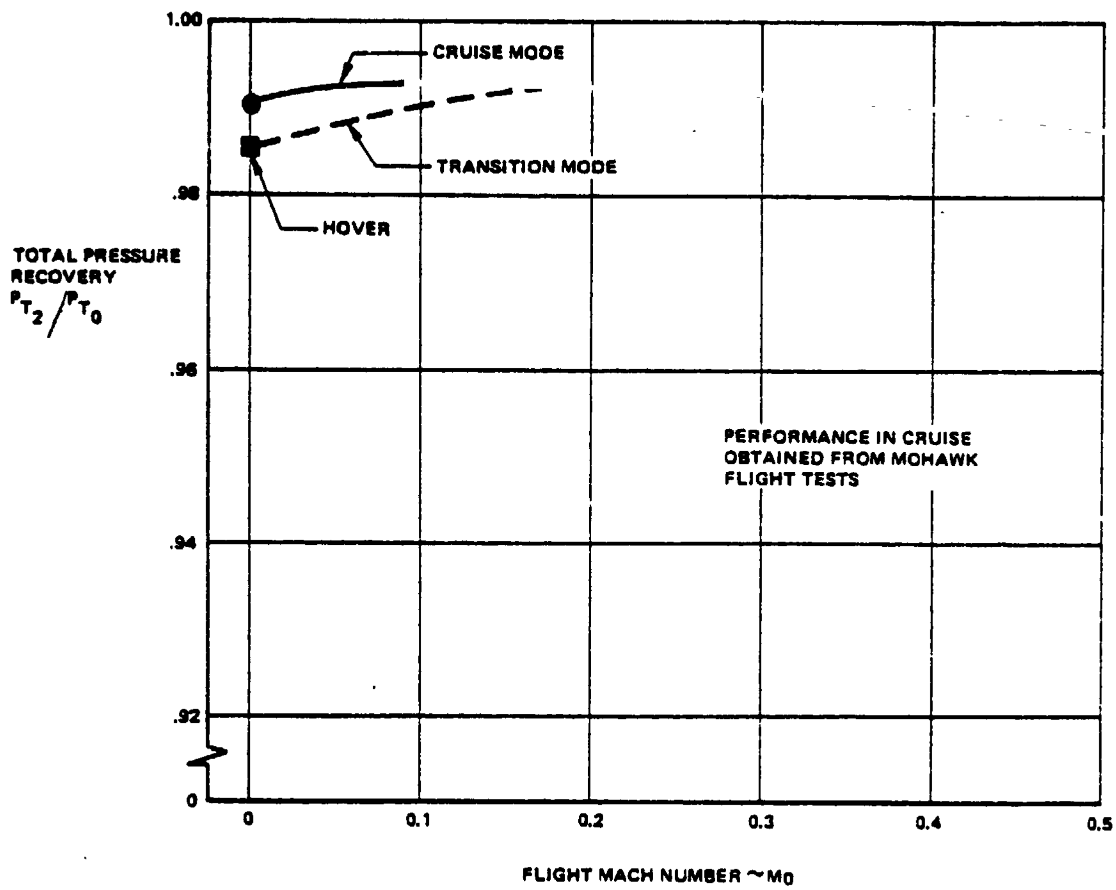
Installed engine performance was calculated using the Lycoming Model Specification No. 104.47 for the LTC 1K-4K turboshaft engine accounting for the following installation effects.

- o Inlet perssure loss.
- o 454 kw (6.95 shp) power extraction from the engine accessory pad.
- o Exhaust tailpipe area 0.11m^2 (164 square inches).

In the hover position the rotors may induce a cross flow at the engine inlets resulting in a small reduction in inlet total pressure recovery. The assumed total pressure recovery variation with Mach number, including estimates for this effect in hover and transition, is presented in Figure 3.2.1.

The tailpipe area of 0.11m^2 (164 square inches) was selected to minimize momentum drag in forward flight at partial power without penalizing hover performance excessively.

HTR XV-15 - ENGINE INLET PERFORMANCE



REF: GAC REPORT NO. FD-134-2 AND 5-001, SECT VIII-D, JULY 30, 1969
AO-1 DEMONSTRATION PROGRESS AND DATA REPORT.

FIGURE 3.2.1. PREDICTED ENGINE INLET PERFORMANCE

Engine ratings are as follows:

Rating		Turbine Inlet Temperature
Contingency	(2 minutes)	1325°K (2385°R)
Takeoff	(10 minutes)	1269°K (2285°R)
Military	(30 minutes)	1236°K (2225°R)
Normal	(continuous)	1205°K (2170°R)

Installed LTC 1K-4K performance is presented in Figures 3.2.2 and 3.2.3.

3.2.2 Airframe Aerodynamics

The aerodynamics of the basic tilting-engine XV-15, as presented in Reference 1, were reviewed and adjustments were made to reflect the effect of the fixed engines on the wing-nacelle-engine lift, drag and pitching moment. No change was estimated in the wing-nacelle lift or pitching moment. Drag was estimated to be unchanged in the cruise configuration, (nacelle angle $i_N = 0°$). The drag coefficient at zero lift with the nacelles up ($i_N = 90°$) was reduced from $C_{D_O} = .212$ to $C_{D_O} = .125$ because of the lower cross sectional area presented by the Boeing nacelle design in helicopter flight.

3.2.3 Hover Performance

The performance in hover of the HTR XV-15 was computed using the installed engine data presented in Section 3.2.1 and the rotor performance predicted in Reference 5. This rotor has been tested in the NASA Ames 40- by 80-foot wind tunnel (Reference 3). An engine-to-rotor power transmission

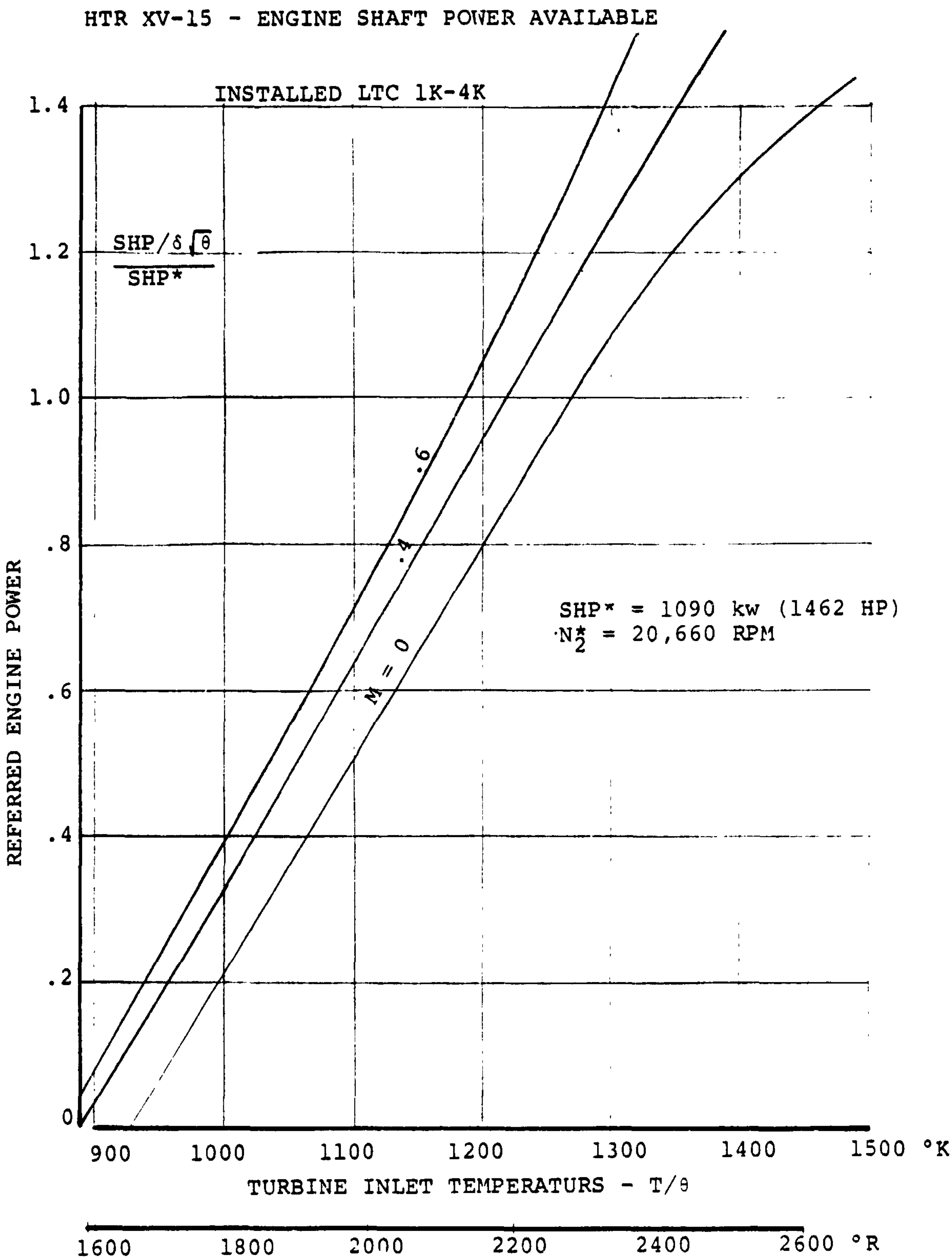


FIGURE 3.2.2. ENGINE SHAFT POWER AVAILABLE

HTR XV-15 - FUEL FLOW CHARACTERISTICS OF INSTALLED
LTC 1K-4K ENGINE

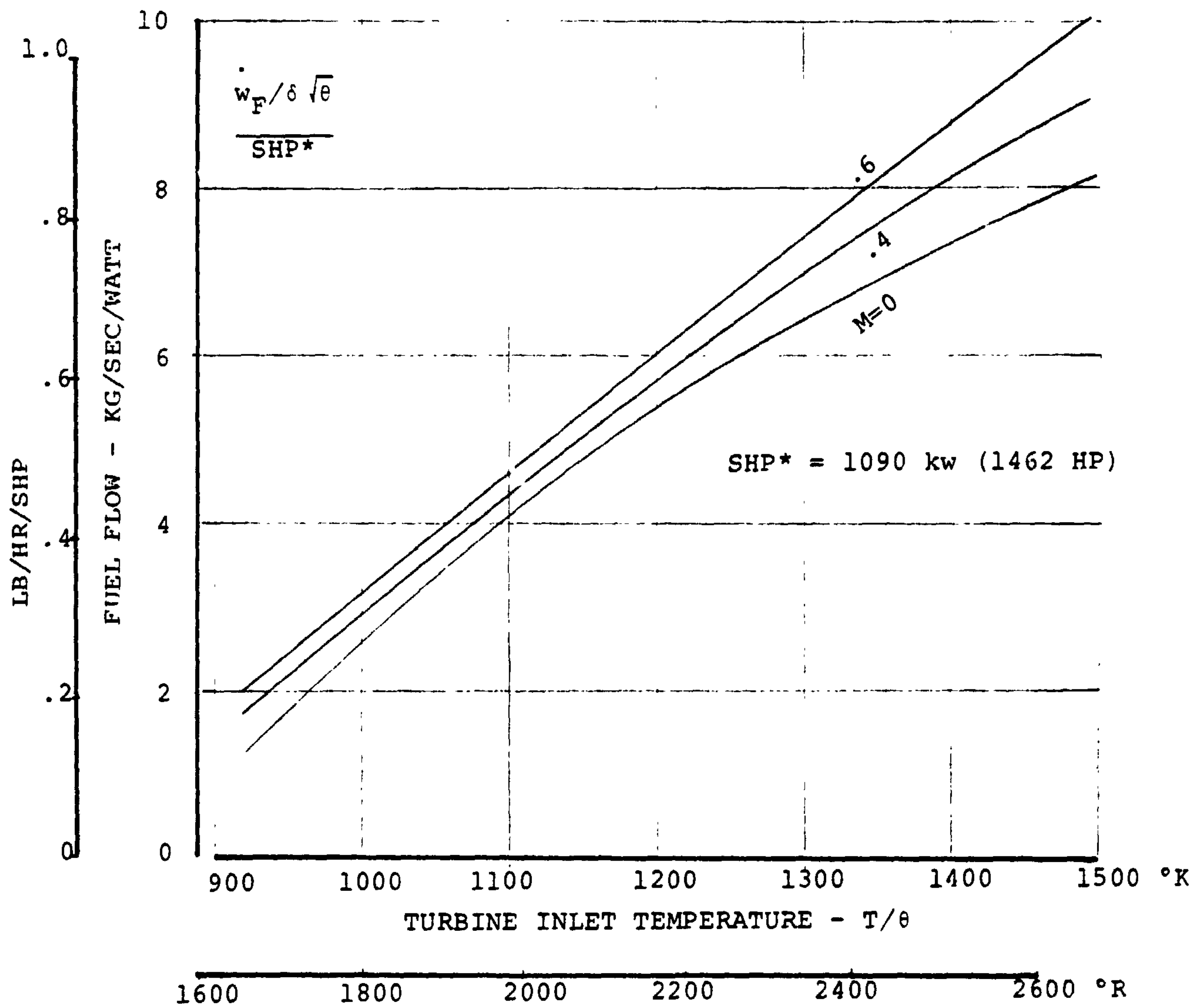


FIGURE 3.2.3. FUEL FLOW CHARACTERISTICS OF INSTALLED ENGINE

efficiency of 0.97 was used, based on an analysis of the gear trains.

The VTOL performance is shown in Figure 3.2.8. in the form of hover ceiling versus gross weight for out-of-ground-effect (OGE) hover with both engines operating (AEO) and with one engine inoperative (OEI). The data is presented for standard day ambient temperatures and for tropical day conditions at two values of lift-to-weight ration, L/W. Lift is defined as the net vertical force available, allowing for download losses. A value of 7% of thrust was assumed for the download experienced out of ground effect. Thus, a value of L/W equal to 1.0 represents a thrust-to-weight ratio, T/W, equal to 1.0753 and for L/W = 1.1, T/W = 1.1828.

A L/W = 1.0 corresponds to maximum hover performance capability as normally defined for helicopters, while L/W = 1.1 provides a 10% margin that can be used for maneuver.

With all engines operating at takeoff power setting, the HTR XV-15 can hover at 805k kg (17,750 lbs) at sea level, standard day or at 6917 kg (15,250 lbs) at sea level, tropical day.

With one engine shut down and the remaining engine operating at contingency power, at sea level, standard day, the aircraft can hover at a gross weight of 5511 kg (12,150 lbs), while at tropical conditions hover weight is 4830 kg (10,650 lbs).

Maximum hover ceiling AEO is 5486 meters (18,000 feet), standard day at the minimum flying weight of 4664 kg (10,284 lbs).

Figure 3.2.4. shows the effect of outside air temperature (OAT) on sea level hover performance. The NASA goal of OGE hover with one engine shut down can be met at a flight gross weight of 5488 kg (12,100 lbs) at sea level standard. At higher temperatures performance is limited by engine power available, while below standard temperature the torque allowable on the cross shaft is limiting.

With all engines operating, hover performance at ambient temperatures greater than standard is again limited by power available. Below this temperature, useable power is limited by the allowable torque on the intermediate shaft.

At design gross weight 6154 kg (13,568 lbs) the aircraft can hover OGE in temperatures up to 43°C (110°F) and at minimum flying weight can hover up to 35°C (96°F).

3.2.4 Transition

The power required in transition is presented in Figure 3.2.5. as a function of airspeed for different nacelle angles. The values shown are for sea level, standard day at the design gross weight of 6154 kg (13,568 lbs). The maximum climb performance available during transition is presented in Figure 3.2.6. The rate of climb is also shown for the condition when fuselage attitude is maintained level.

3.2.5 Cruise

The cruise performance of the HTR XV-15 at the design gross weight is shown in Figure 3.2.7 With all engines operating the maximum speed is limited by transmission torque up to an altitude of 5.2 km (17,000 feet) where a true airspeed of 324 knots is attainable. With one engine inoperative, torque is limiting up to 1 km (3,300 feet). Beyond this altitude, at normal rated power 212 knots can be attained at 3 km (10,000 feet).

HTR XV-15 - EFFECT OF AMBIENT TEMPERATURE ON HOVER PERFORMANCE

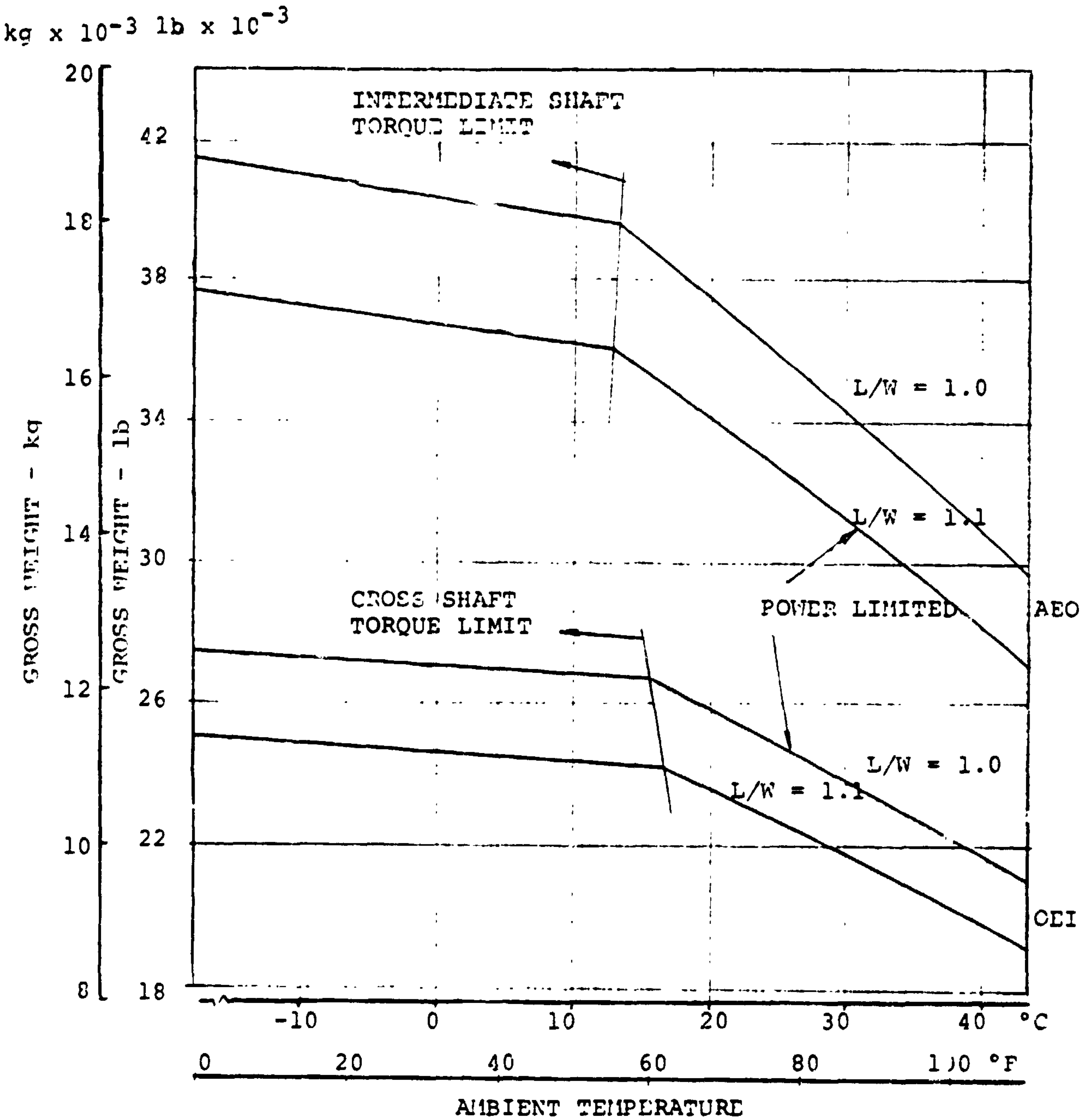


FIGURE 3.2.4. EFFECT OF AMBIENT TEMPERATURE ON HOVER PERFORMANCE - SEA LEVEL

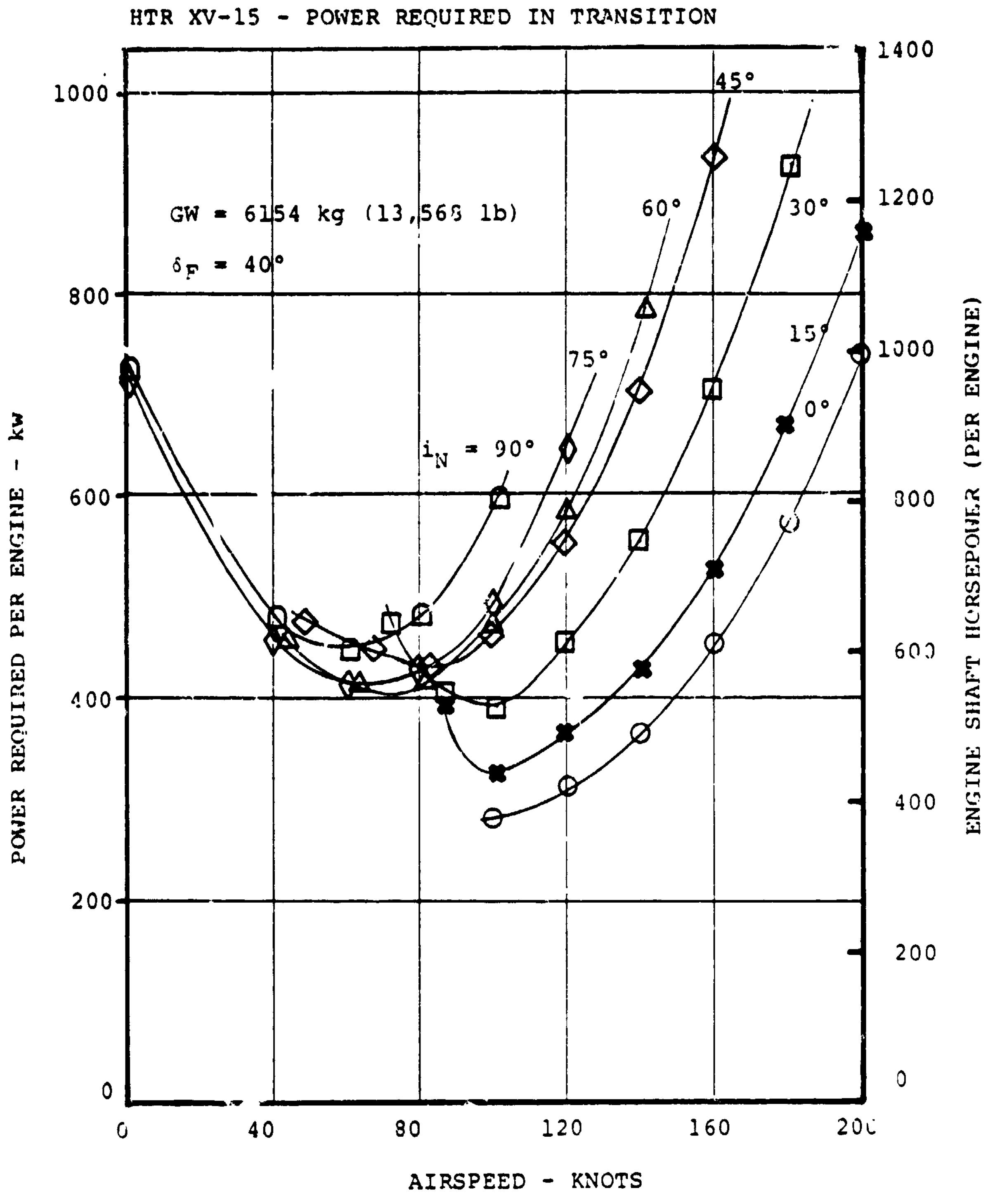


FIGURE 3.2.5. POWER REQUIRED IN TRANSITION - AFT CG, SEA LEVEL

HTR XV-15 - CLIMB PERFORMANCE THROUGH TRANSITION

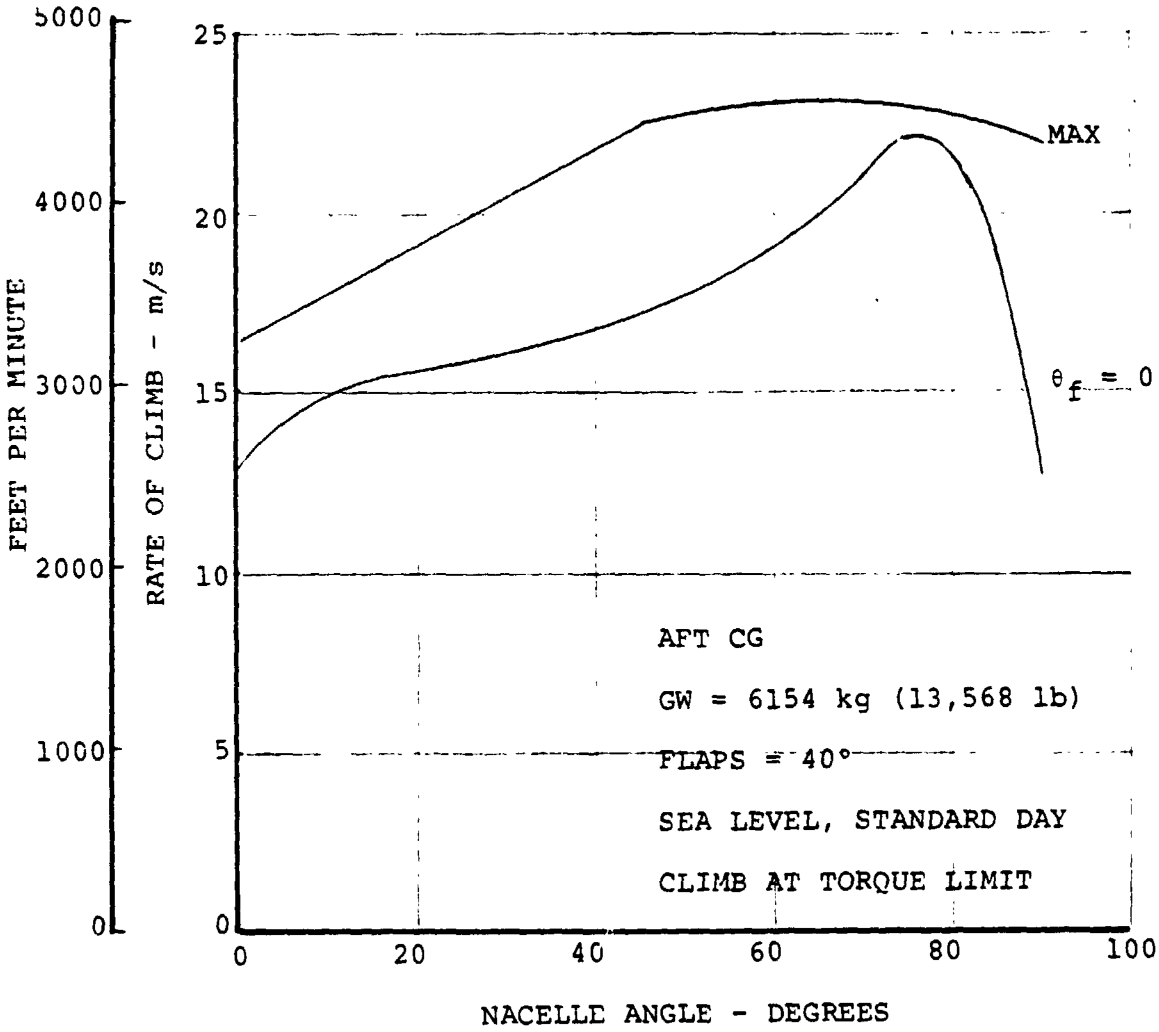


FIGURE 3.2.6. RATE-OF-CLIMB PERFORMANCE IN TRANSITION

C-3

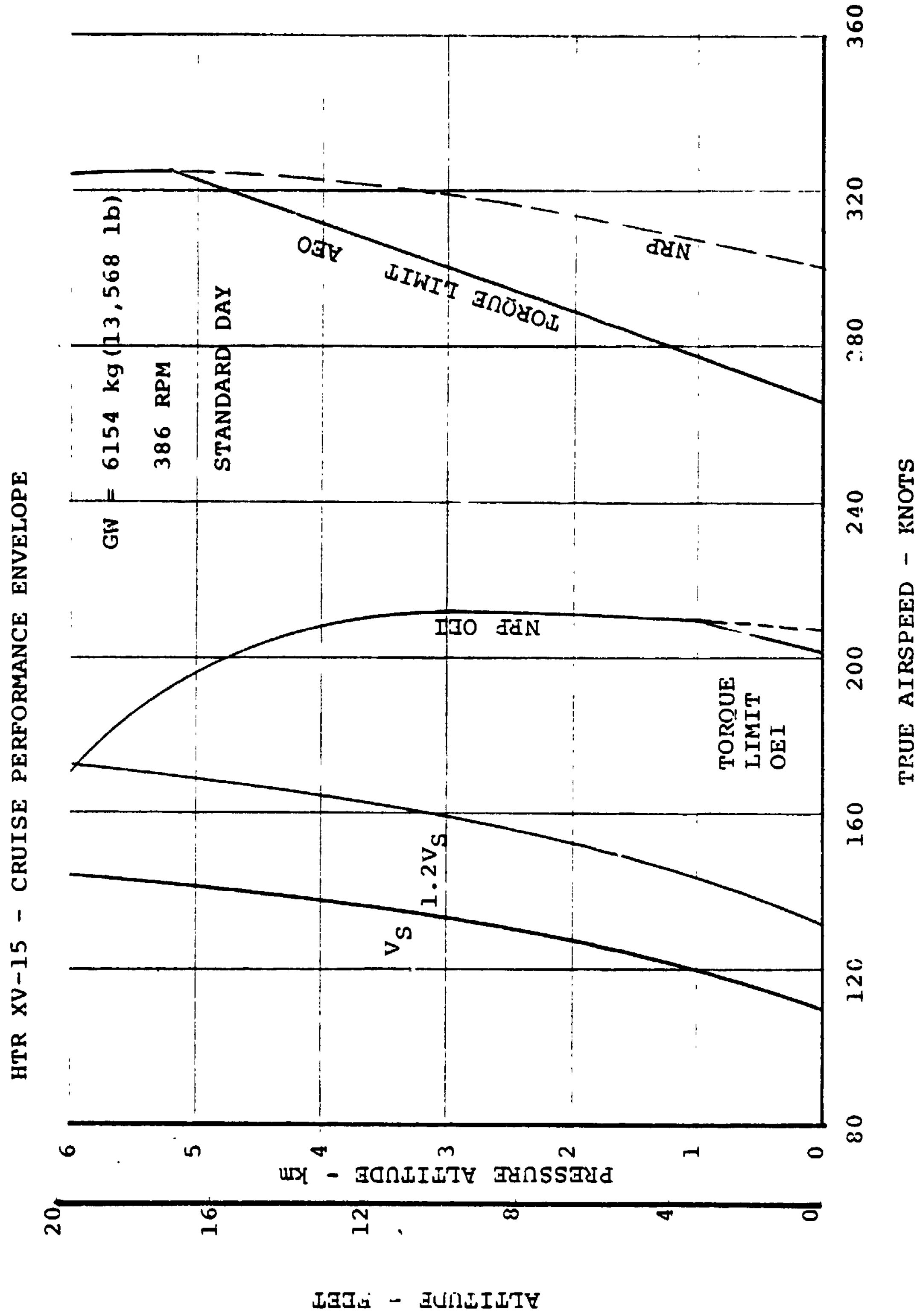


FIGURE 3.7.7 CRUISE PERFORMANCE ENVELOPE

HTR XV-15 - HOVER CEILING VERSUS GROSS WEIGHT

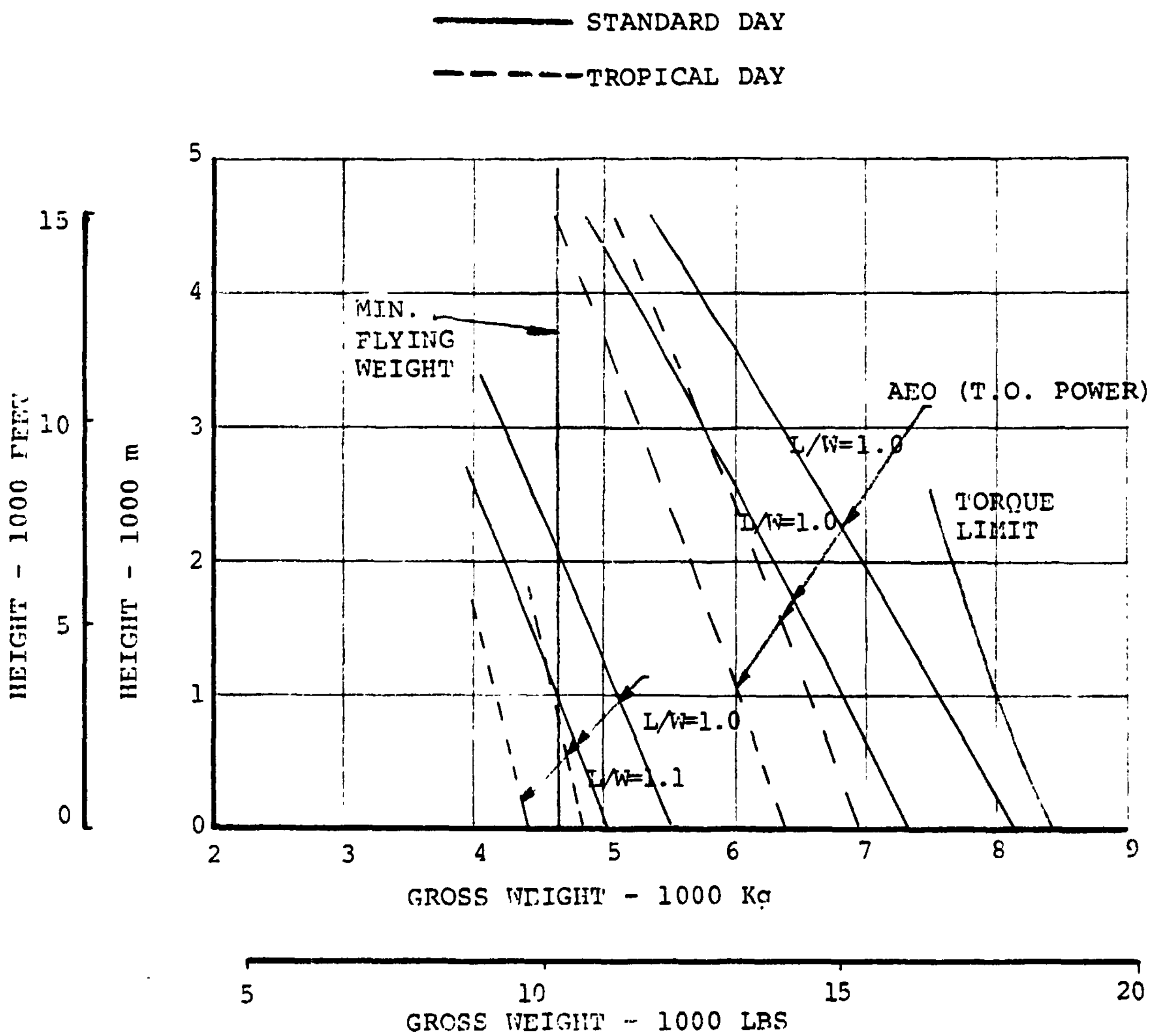


FIGURE 3.2.8. HOVER CEILING VERSUS GROSS WEIGHT

lbs). With one engine inoperative this is reduced to 2011m (6,600 feet). The corresponding tropical day performance is 4298m (14,100 feet) AEO and 762m (2,500 feet) OEI. As noted on the figure, the hover performance is not torque limited.

The rate-of-climb capability at normal rated power is presented in Figure 3.2.9 for three different gross weights. Below 3 km (10,000 feet) transmission torque limits the rate of climb. At the design gross weight sea level, rate of climb is estimated to be 16.5 m/s (3,250 ft/min.).

3.2.6 Mission Performance

Payload versus range capability of the HTR XV-15 is shown in Figure 3.2.10 at the design gross weight and at 7258 kg (16,000 lbs). The data is for a 6096m (20,000 feet) cruise altitude. Maximum range at the design takeoff weight is approximately 611 km (330 nautical miles). Allowance was made for a 10% fuel reserve.

The generalized mission endurance of the aircraft is presented in Figure 3.2.11 as a plot of hover time versus loiter time for various cruise times. The lines of constant cruise time include the time required to climb to the 20,000 feet cruise altitude. The definition of the mission is as follows:

1. Hover at L/W = 1.0 for 1/2 hover time at sea level.
2. Climb to 6096m (20,000 feet).
3. Cruise at 99% best range speed for 1/2 cruise time.

HTR XV-15 - RATE OF CLIMB AS A FUNCTION OF ALTITUDE

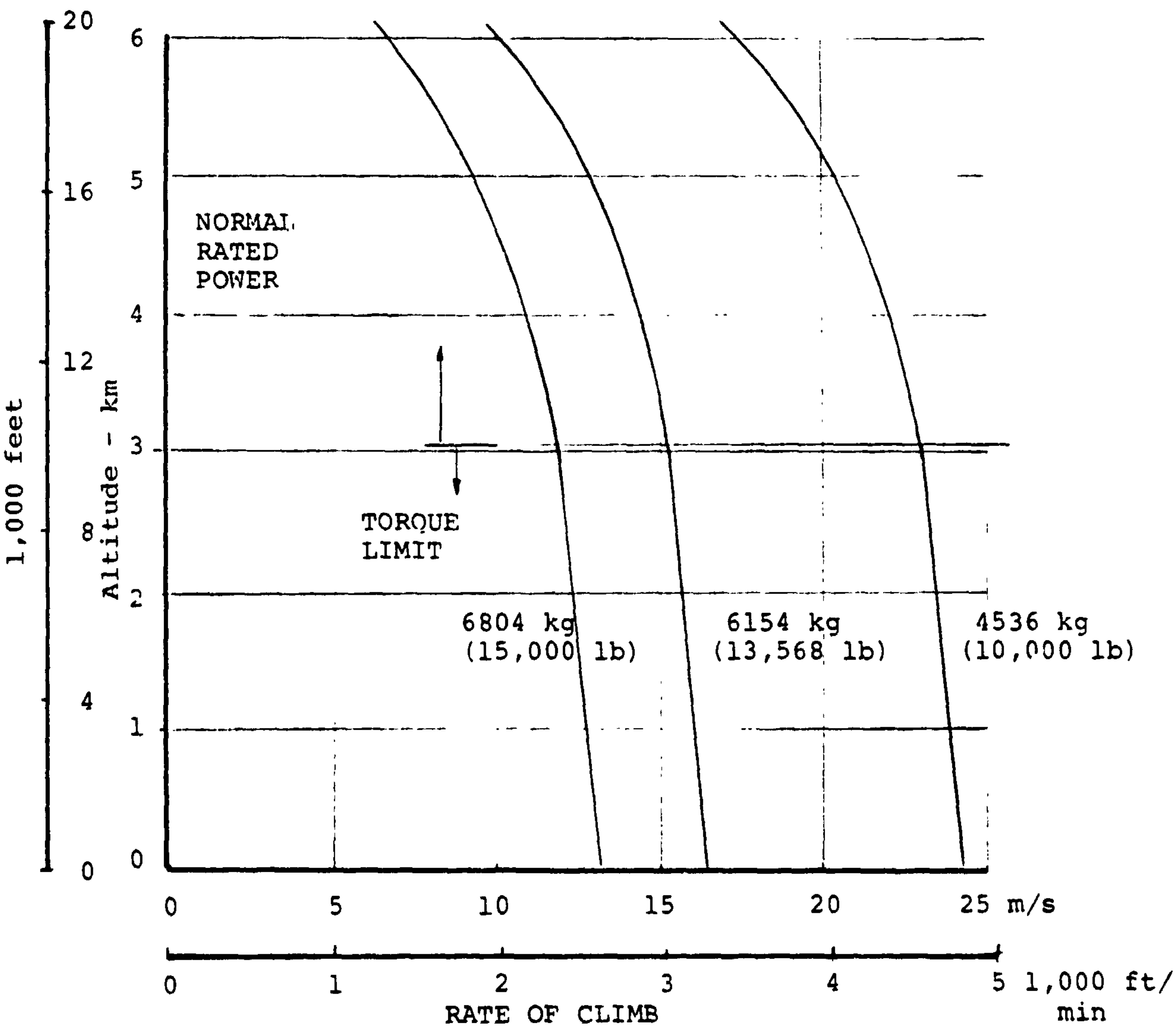


FIGURE 3.2.9. RATE OF CLIMB VERSUS ALTITUDE, AEO

HTR XV-15 - PAYLOAD RANGE CHARACTERISTICS

MISSION: (1) T.O. 2 MIN. AT MAX POWER, HOVER OGE AT L/W = 1
(2) CLIMB TO 20,000 FEET MAX POWER
(3) CRUISE AT $V_{.99BR}$
(4) RESERVE = 10% INITIAL FUEL

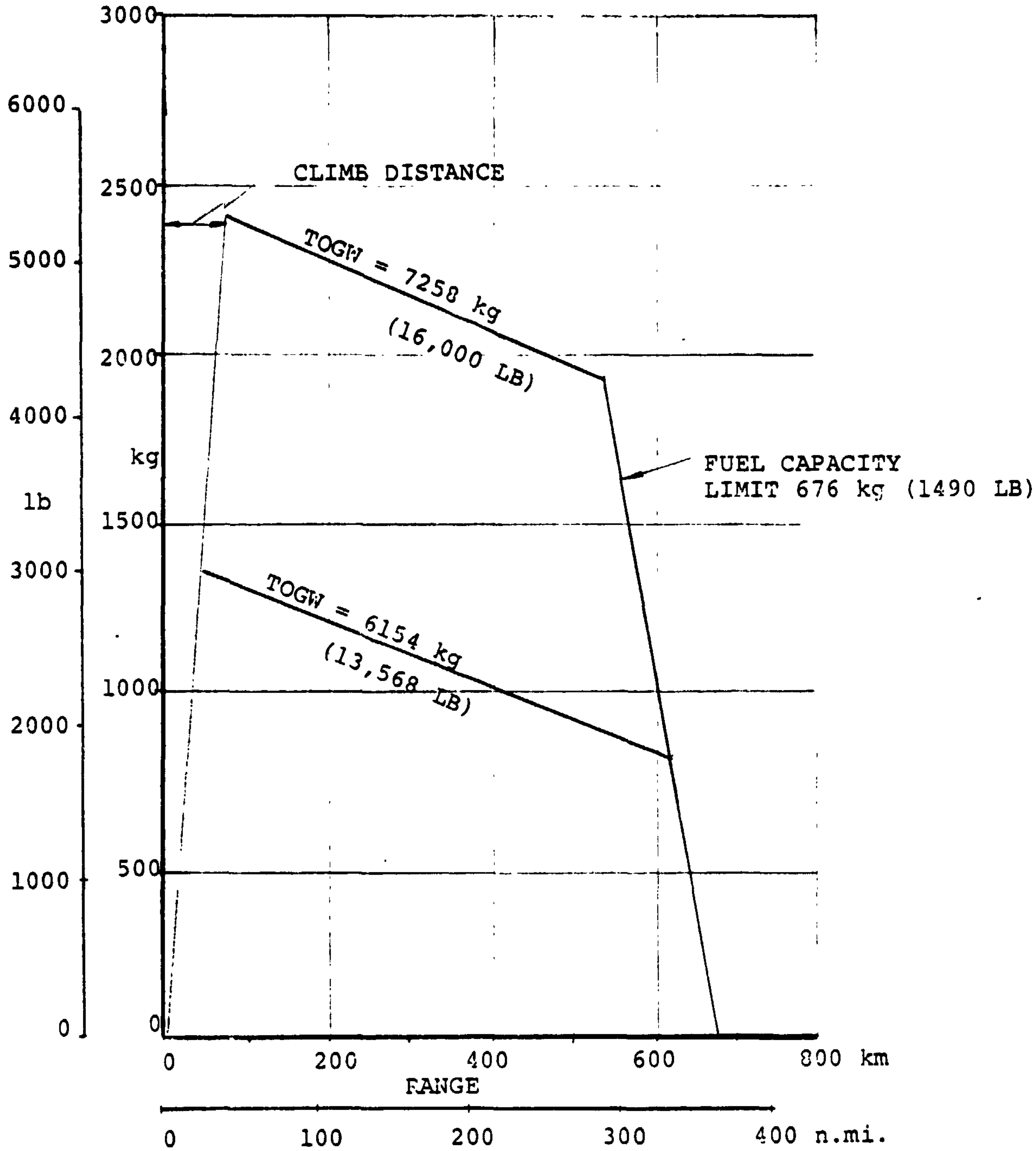


FIGURE 3.2.10. HTR XV-15: PAYLOAD VERSUS RANGE

HTR XV-15 - GENERALIZED ENDURANCE AVAILABLE

TAKEOFF GROSS WEIGHT = 13,568 LBS
LANDING GROSS WEIGHT = 12,227 LBS

USEABLE FUEL = 1,341
RESERVE = 149
1,490

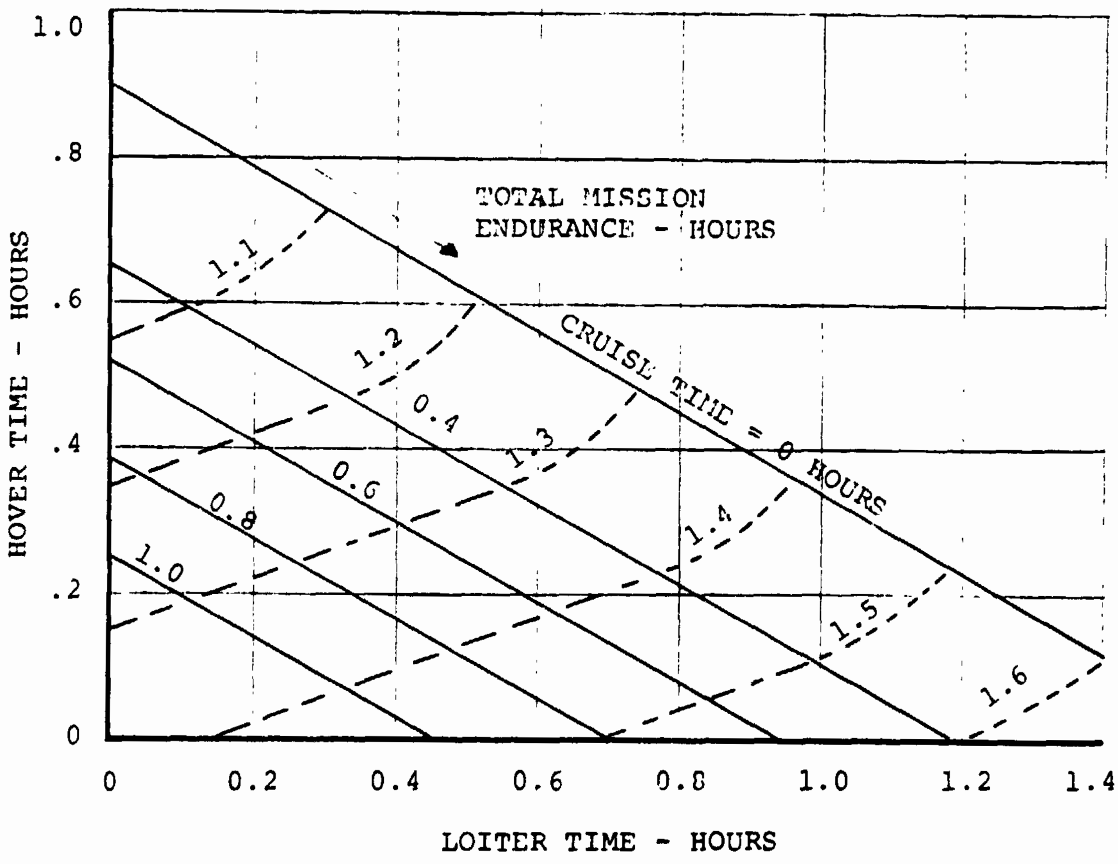


FIGURE 3.2.11. GENERALIZED ENDURANCE CAPABILITY

4. Transfer to 3048m (10,000 feet).
5. Loiter at best endurance speed.
6. Transfer to 6096m (20,000 feet).
7. Cruise for 1/2 cruise time at best range speed.
8. Transfer to sea level.
9. Hover for 1/2 hover time at sea level at L/W = 1.0.

For cruise times greater than 0.4 hours the cruise includes fuel and time for climb to 20,000 feet. For cruise times less than 0.4 hours the fuel and time to climb to 10,000 feet is included. The calculations were made for a takeoff weight of 6154 kg (13,568 lbs) - the design gross weight - and include a fuel reserve of 10% of initial fuel. The generalized mission performance plot shows, for example, that a total mission time of 1.5 hours can comprise either one hour of loiter, 0.4 hours of cruise and 0.1 hours of hover; or, zero hover time with 0.7 hours of loiter and 0.8 hours of cruise.

3.2.7 Comparison With Existing XV-15 Performance

Some comparisons of the performance of the fixed-engine HTR XV-15 with that published for the XV-15 are presented in Table 3.2.1. Data for the XV-15 performance was obtained from Bell Helicopter Company Report 301-199-001, Revision A. The comparison is based on the assumption that both aircraft are required to carry the same useful load.

TABLE 3.2.1. PERFORMANCE COMPARISON

	XV-15		HTR XV-15	
Design Gross Weight, kg (lb)	5896	(13000)	6154	(13568)
Weight Empty kg (lb)	4117	(9076)	4374	(9644)
Useful Load kg (lb)	1780	(3924)	1780	(3924)
Max. Level Flight Speed at Torque Limit, Knots				
Sea Level Standard Day	260		266	
6096m (20000 ft) at NRP	304		325	
Hover Endurance, Hours	0.79		0.9	
Max. Rate of Climb at Torque Limit, m/s (ft/min)				
Helicopter $i_N = 75°$	16.0	(3150)	22.85	(4500)
Airplane $i_N = 0°$	14.6	(2875)	16.5	(3250)
Hover Ceiling m (ft) Std Day, OGE, T.O. Power	2835	(9300)	3322	(10900)

3.3 NOISE ASSESSMENT

3.3.1 Methodology for Far Field and Near Field Noise Prediction

Far-field (greater than one rotor diameter from blade tips) noise for the Boeing Vertol Hingeless Rotor variant of the XV-15 tilt rotor aircraft were assessed, using the Graphical Prediction Methods of the FAA Report FAA-RD-76-49, II (V/STOL Rotary Propulsion Systems Noise Prediction and Reduction). The propeller noise method of the report was selected in preference to helicopter rotor noise calculation methods, since the tilt rotor with its greater blade aerodynamic twist has a spanwise airload distribution which more closely resembles a free air propeller than a helicopter rotor. The prediction includes the combined effects of steady loading noise, unsteady loading noise, and broadband noise, and calculates the frequency spectrum as well as the perceived noise level (PNL). Near-field (less than one rotor diameter from blade tips) noise was estimated using the prediction procedure for propellers contained in the SAE Aerospace Information Report 1407 (5/77). This method calculates the noise on the fuselage surface using input parameters of propeller-diameter, tipspeed, power input, and relative location to the fuselage.

3.3.2 Hover Noise

Figure 3.3.1 shows the predicted perceived noise level (PNL) for a range of tipspeeds and gross weights for a distance of 500 feet from the prop/rotors. Also noted on the figure is the PNL for the design condition. The figure illustrates noise sensitivity to tipspeed and gross weight, and shows that over a range of gross weights from 5591 kg (12300 lb) to 7091 kg (15600 lb) the noise level increases from 88.5 PNdb to 91 PNdb. The perceived noise level is seen to be proportional to tipspeed. However, noise reduction resulting from a reduced tipspeed would penalize hover performance.

3.3.3 Cruise Noise

During a cruise condition, as shown in Figure 3.3.2, the tilt prop/rotor provides an excellent configuration for minimum noise exposure due to its low tipspeed (525 feet per second), and lightly loaded rotor (1.36 lbs/ft^2 - with the 26-foot diameter rotor).

A PNL comparison of three cruise airspeeds (160, 200 and 250 knots) is shown for a ground point 1,000 feet below the flight path. As indicated in the figure, the greater airspeeds cause an increase in the maximum levels perceived by the observer for shorter time durations.

At a normal cruise speed of 200 knots, the maximum PNL does not exceed 61 dB and increases above 50 dB for approximately only ten seconds.

3.3.4 Noise at the Fuselage

The near-field noise on the rotor, during cruise at the nearest point on the fuselage relative to the prop/rotor, is shown in Figure 3.3.3. These results show that noise pressure level is highly sensitive to the separation between fuselage and blade tip, increasing sharply as the distance is reduced. By

HTR XV-15 ROTOR NOISE - HOVER, 500 FT. DISTANCE

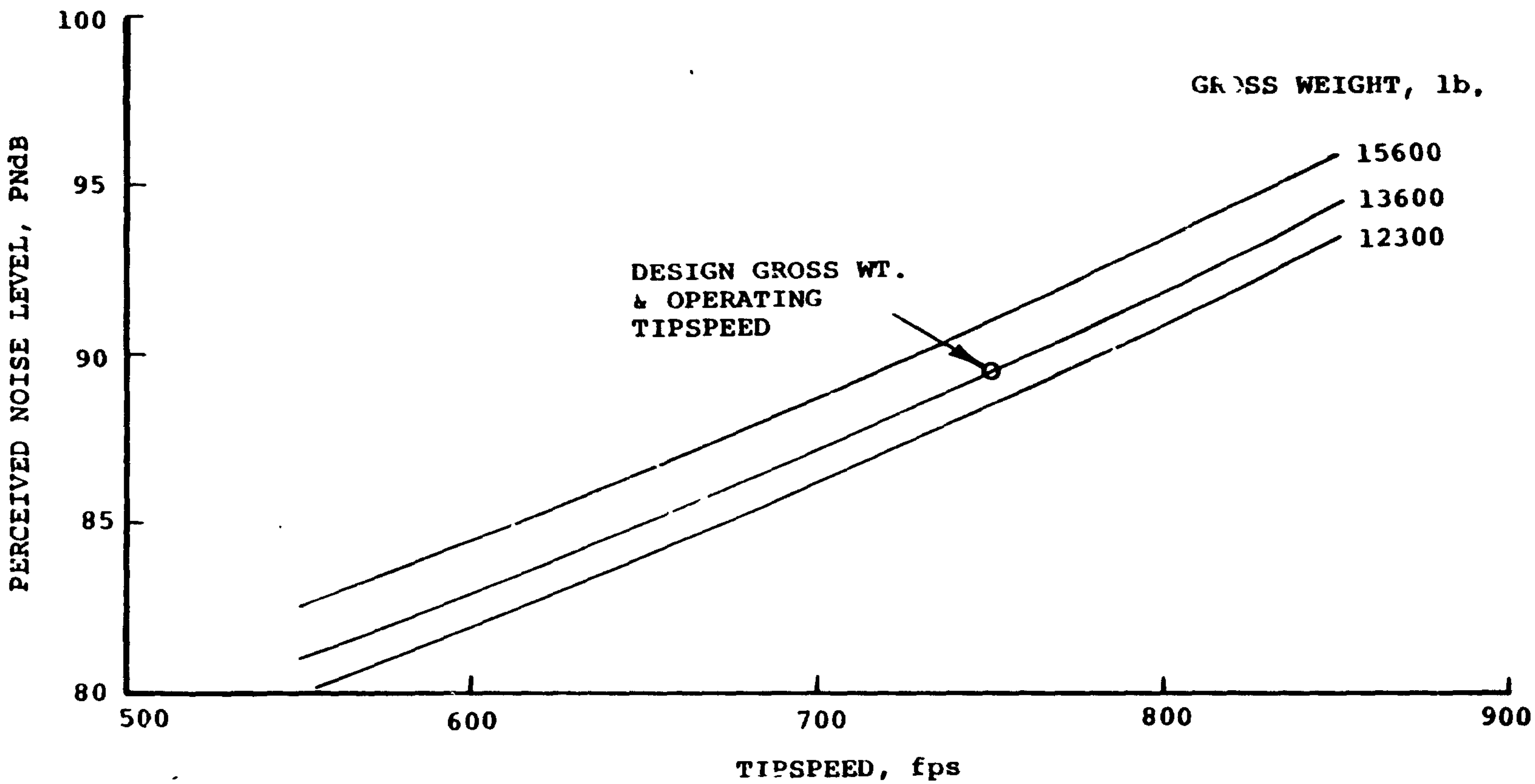


FIGURE 3.3.1 XV-15 HOVER ROTOR NOISE AT 500 FT.

HTR XV-15 ROTOR NOISE TIME HISTORY

DURING LEVEL CRUISE FLYOVER - 1000 FT. ABOVE OBSERVER

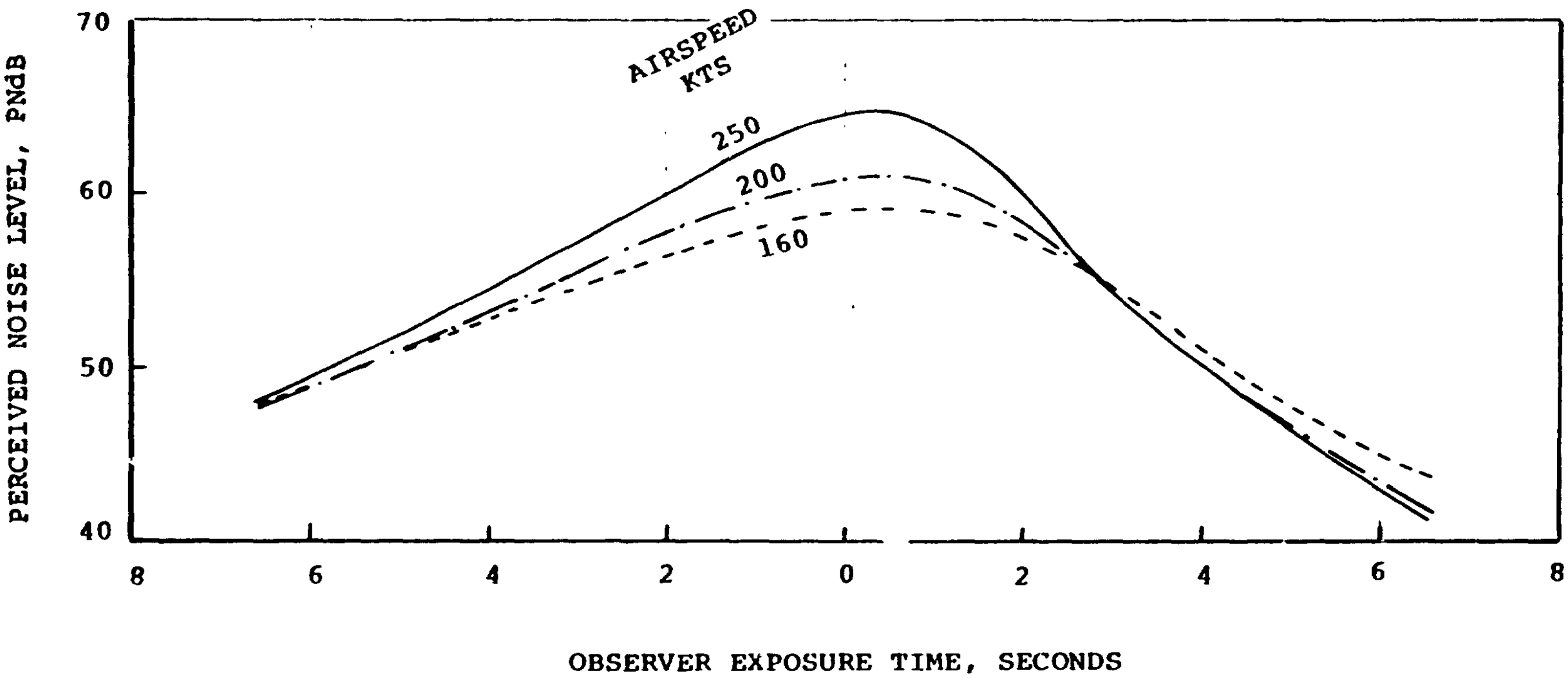


FIGURE 3.3.2 HTR-XV15 FLY-BY NOISE, 1000 FT.

HTR XV-15 ROTOR NOISE AT THE FUSELAGE SURFACE

IN THE PLANE OF ROTATION DURING A

LEVEL CRUISE FLIGHT CONDITION

200 kts, 525 ft/sec TIPSPEED

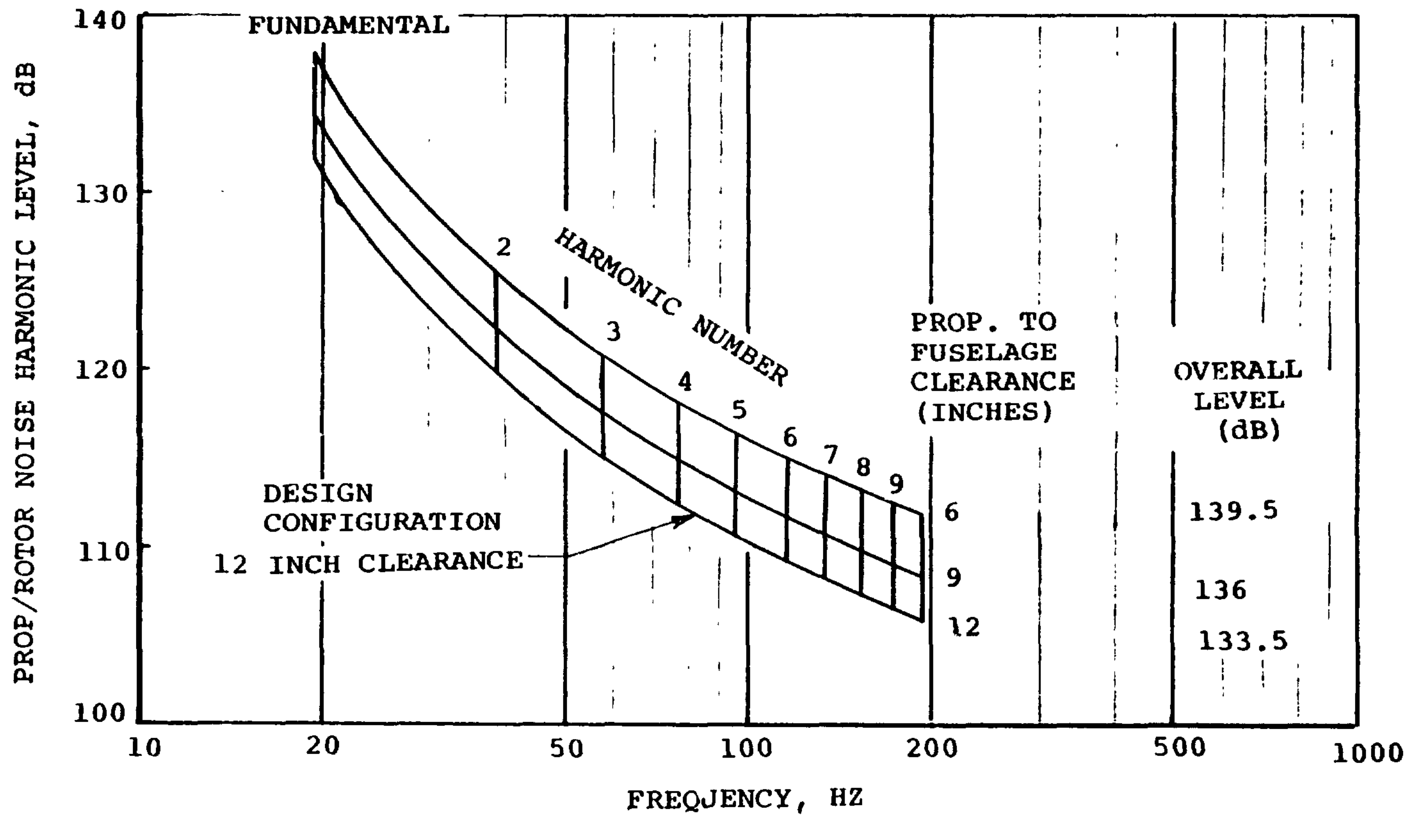


FIGURE 3.3.3 FREQUENCY SPECTP'JM OF HTR XV-15, FUSELAGE SOUND PRESSURE LEVELS FOR THREE VALUES OF TIP CLEARANCE

providing a 12-inch rotor tip-to-fuselage minimum clearance, and a low tipspeed (525 feet per second), the predicted overall sound pressure level is 133.5 dB. This is substantially below the criterion level of 140 dB level quoted in MIL-A-8893 (USAF), "Airplane Strength and Rigidity; Sonic Fatigue," for discrete frequency noise or prop/rotor harmonic noise, for which areas of the fuselage are to be considered susceptible to sonic fatigue.

3.4 AEROELASTIC STABILITY

3.4.1 Methodology

The HTR XV-15 was evaluated using the methodology of NASA TN D-8515. This methodology was developed by Dr. Wayne Johnson of the Large Scale Aerodynamics Branch of the Ames Research Center and in several respects improves on methods in previous studies at Boeing Vertol. The principal distinguishing feature of this methodology is the relative ease with which the effect of trim blade deflections may be considered. These introduce coupling between blade pitch/torsion and deflection parallel (lead-lag) and normal (flap) to the plane of the rotor. Other important features include initial calculation of trim so that realistic flight conditions are investigated. The aerodynamic analysis deals with nonaxial flow (high Aq) and periodic coefficients may be optionally generated and included in the solution if they are believed to affect stability. Because of the relatively low values of μ encountered in tilt rotor operation, periodic coefficients are ignored in the present study.

3.4.2 Mathematical Model

The aircraft representation includes the rigid body degrees of freedom, four flexible modes of the structure, and the fundamental flexure modes of the blade. Blade torsion is represented by two modes, a rigid blade pitch against the control linkage stiffness and a fundamental cantilevered torsion mode. Also included in the model are the rotor rotational degree of freedom, transmission dynamics, axial inflow perturbations, and rpm coupling of the rotor governor.

Symmetric and anti-symmetric degrees of freedom are analyzed separately.

3.4.3 Aircraft Data

Blade structural design data is given in Table 3.4.1. This is a twin-pin retention blade designated as Design 37. Other data is given in Appendix II. The blade rotating natural frequencies are given in Table 3.4.2 and the carpet plots of the fundamental flexural frequencies are given in Figures 3.4.1 and 3.4.2.

Airframe

For this calculation, airframe frequencies were estimated by ratioing published Bell data for the XV-15 to account for changes in the tip package mass and engine location. (NOTE: Subsequent NASTRAN vibration analyses of the airframe has indicated that additional detail design in the nacelle region will be necessary to ensure acceptable frequency and mode shape characteristics. See structural evaluation, Section 3.5). Blades-off natural frequencies used in the stability analysis are given in Table 3.4.3.

3.4.4 Results

3.4.4.1 High Speed Cruise

As shown in Figure 3.4.3, the system develops an instability at 360 knots. There is a reduction in critical speed as rpm increases, but the deterioration is gradual. This represents

TABLE 3.4.i

HTR-XV15 ROOT END
SUMMARY OF SECTION PROPERTIES AND
FATIGUE MARGINS

DESIGN 37

r/R	WT LB/IN (1)	AE LB x 10^{-6}	EI_F LB-IN^2 x 10^{-6}	EI_C LB-IN^2 x 10^{-6}	GJ LB-IN^2 x 10^{-6}	I_p LB IN^2/IN (2)	FATIGUE M.S. (3)
0	7.2		1 x 10	1 x 10			
.017	7.2		1 x 10	1 x 10			
.017	7.2		580.8	680.7			
.071	7.2		580.8	680.7			
.071	.982	55.8	580.8	680.7	87.8	2.13	+.10
.090			580.8	680.7	87.8		
.10	.688	47.0	72.7	144.8	53.4	1.54	+.88
.15	.438	30.7	22.7	46.6	22.8	.91	+.10
.20	.349	24.4	12.0	31.9	15.4	.97	+.03
.25	.283	18.4	9.5	29.1	15.8	.80	+.01
.30	.251	14.5	7.1	30.8	15.6	.84	-.04
.45	.456	12.36	4.96	495.0	11.3	12.7	
.50	.442	11.4	4.38	475.0	11.3	12.7	
.60	.434	10.3	3.77	465.0	11.0	12.7	
.70	.429	10.1	3.67	470.0	10.8	12.62	
.75	.426	10.0	3.48	470.0	11.8	12.60	
.80	.425	9.8	3.30	475.0	12.8	13.59	
.85	.424	9.9	3.31	482.0	14.0	12.53	
.90	.415	10.1	3.32	490.0	15.3	12.51	
.95	.415	10.0	3.29	490.0	15.7	12.51	
.975	.415	10.0	3.26	490.0	15.9	12.51	
.975	.415	10.0	3.26	490.0	15.9	12.51	
1.0	.415	9.9	3.24	490.0	16.2	12.51	

(1) BALANCE WEIGHTS NOT INCLUDED IN LB/IN. TABULATION. THESE ARE 1.33 LBS AT .7725, .80 AND .8225 r/R (i.e., NOMINALLY 4 LBS AT .80 r/R). TIP WEIGHT IS 4.2 LBS SPREAD OVER .9775 TO 1.0 r/R.

(2) VALUES FROM DESIGN 29A; VALUES FOR DESIGN 37 WILL NOT BE SIGNIFICANTLY DIFFERENT.

(3) COMPOSITE FATIGUE M.S. WITHOUT SHEAR STRAINS. EFFECT WAS JUDGED TO BE INSIGNIFICANT BASED ON CHECK MADE DURING DESIGN 31 AT r/R = .2 AND .25. FIGURES FROM 0.30 TO TIP NOT AVAILABLE.

TABLE 3.4.2

HTR-XV15 DESIGN 37 BLADE:
NATURAL FREQUENCIES VS RPM AND COLLECTIVE PITCH

ROTOR RPM	MODE	ω/Ω VERSUS COLLECTIVE PITCH AND RPM			
		9°	25°	40°	60°
125	1	1.551	1.463	1.376	1.291
	2	2.642	2.691	2.735	2.776
	3	6.758	6.744	6.736	6.702
	4	15.62	15.61	15.59	15.57
	5	16.29	16.28	16.28	16.27
250	1	1.154	1.043	.953	.876
	2	1.534	1.611	1.664	1.705
	3	4.238	4.216	4.186	4.148
	4	7.854	7.834	7.805	7.763
	5	9.101	9.092	9.080	9.066
386	1	.909	.837	.778	.731
	2	1.299	1.346	1.380	1.405
	3	3.540	3.513	3.477	3.431
	4	5.139	5.110	5.065	4.999
	5	6.870	6.857	6.843	6.825
551	1	.747	.706	672	.647
	2	1.219	1.242	1.260	1.273
	3	3.232	3.202	3.163	3.112
	4	3.665	3.623	3.559	3.466
	5	5.797	5.783	5.767	5.749
750	1	.648	.625	.607	.595
	2	1.180	1.192	1.201	1.206
	3	2.771	2.715	2.629	2.501
	4	3.072	3.040	2.998	2.945
	5	-	-	-	-

HTR XV-15 BLADE DESIGN NO. 37
FIRST FLEXURAL MODE FREQUENCY VARIATION
WITH COLLECTIVE AND RPM

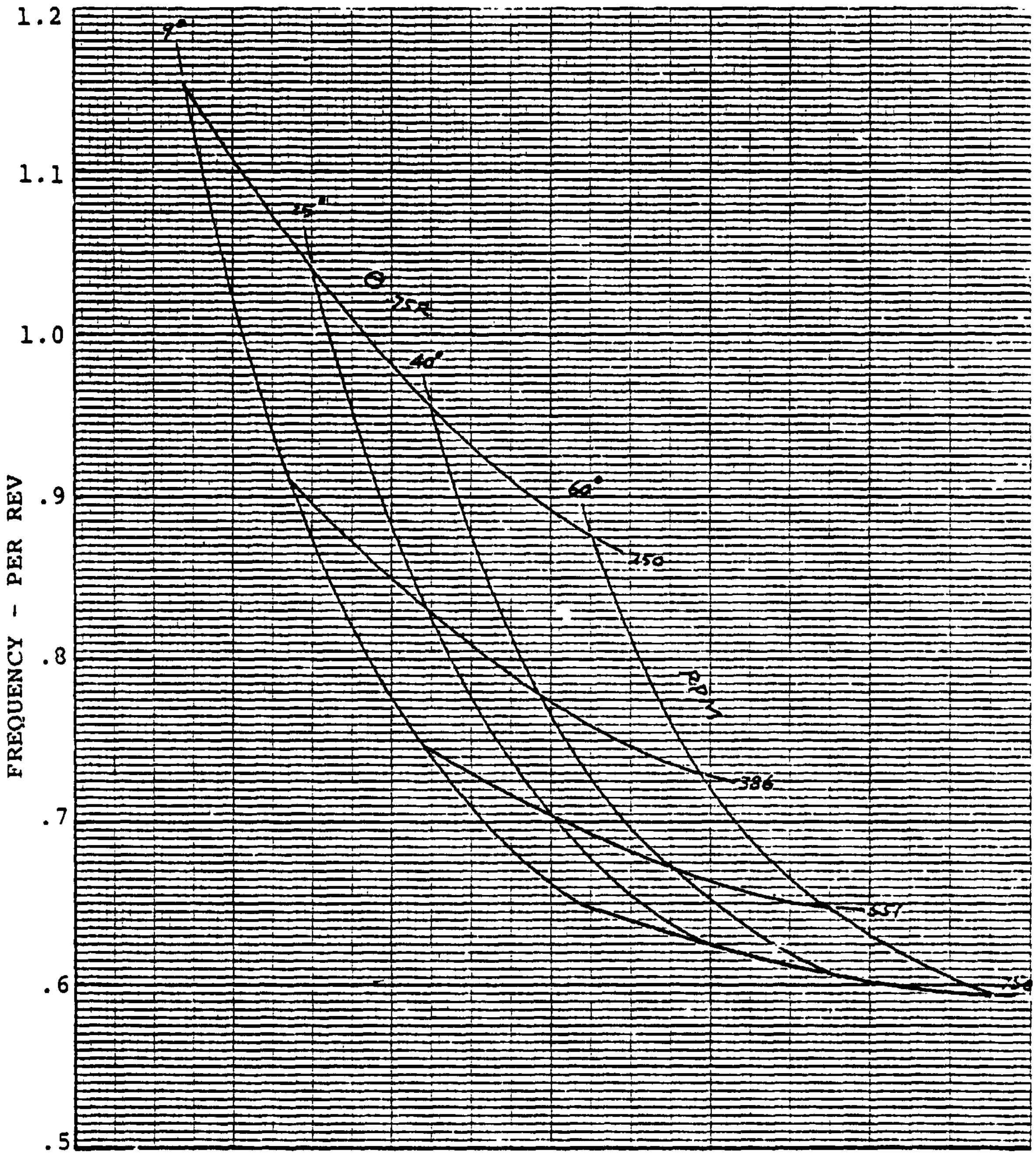


FIGURE 3.4.1. PLOT OF FIRST FLEXURE FREQUENCY AS FUNCTION OF COLLECTIVE AND RPM

HTR XV-15 BLADE DESIGN NO. 37
SECOND FLEXURAL MODE FREQUENCY VARIATION
WITH COLLECTIVE AND RPM

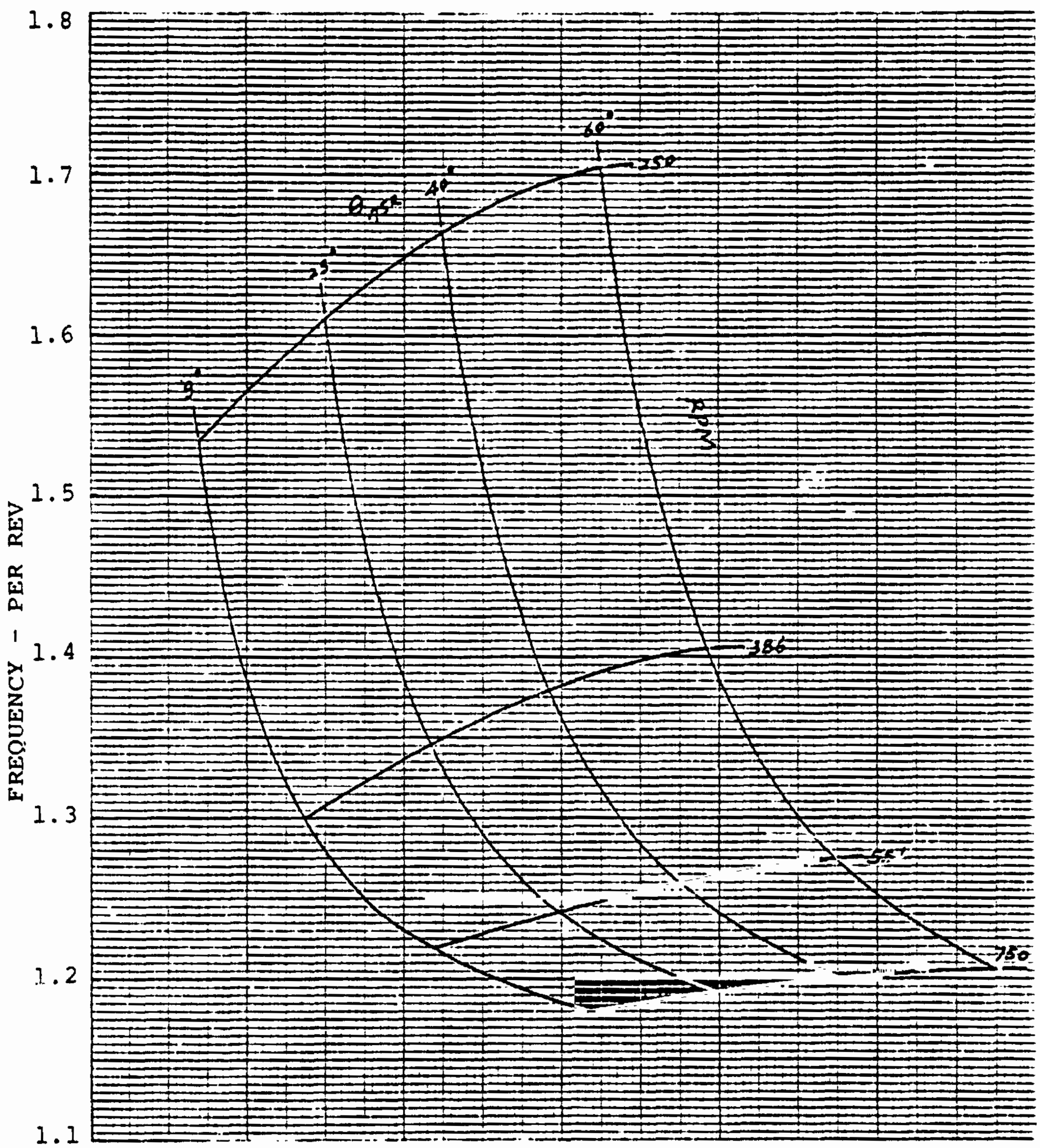


FIGURE 3.4.2. PLOT OF SECOND FLEXURE FREQUENCY AS FUNCTION OF COLLECTIVE AND RPM

TABLE 3.4.3. AIRFRAME MODAL FREQUENCIES USED IN STABILITY ANALYSIS

i_N DEGREES	VERTICAL BENDING	CHORD BENDING	TORSION	PYLON
SYMMETRIC				
0 Locked	3.109	6.34	8.16	19.7
0 Unlocked	3.107	5.56	6.83	13.8
30	3.085	6.16	6.86	11.7
60	3.038	5.97	7.05	11.1
90	3.014	5.236	7.18	10.9
ANTI-SYMMETRIC				
0 Locked	6.74	8.72	7.46	22.8
0 Unlocked	7.45	6.87	6.06	11.6
30	7.36	7.27	6.41	10.9
60	7.62	7.21	6.04	10.3
90	7.75	7.14	5.33	10.1

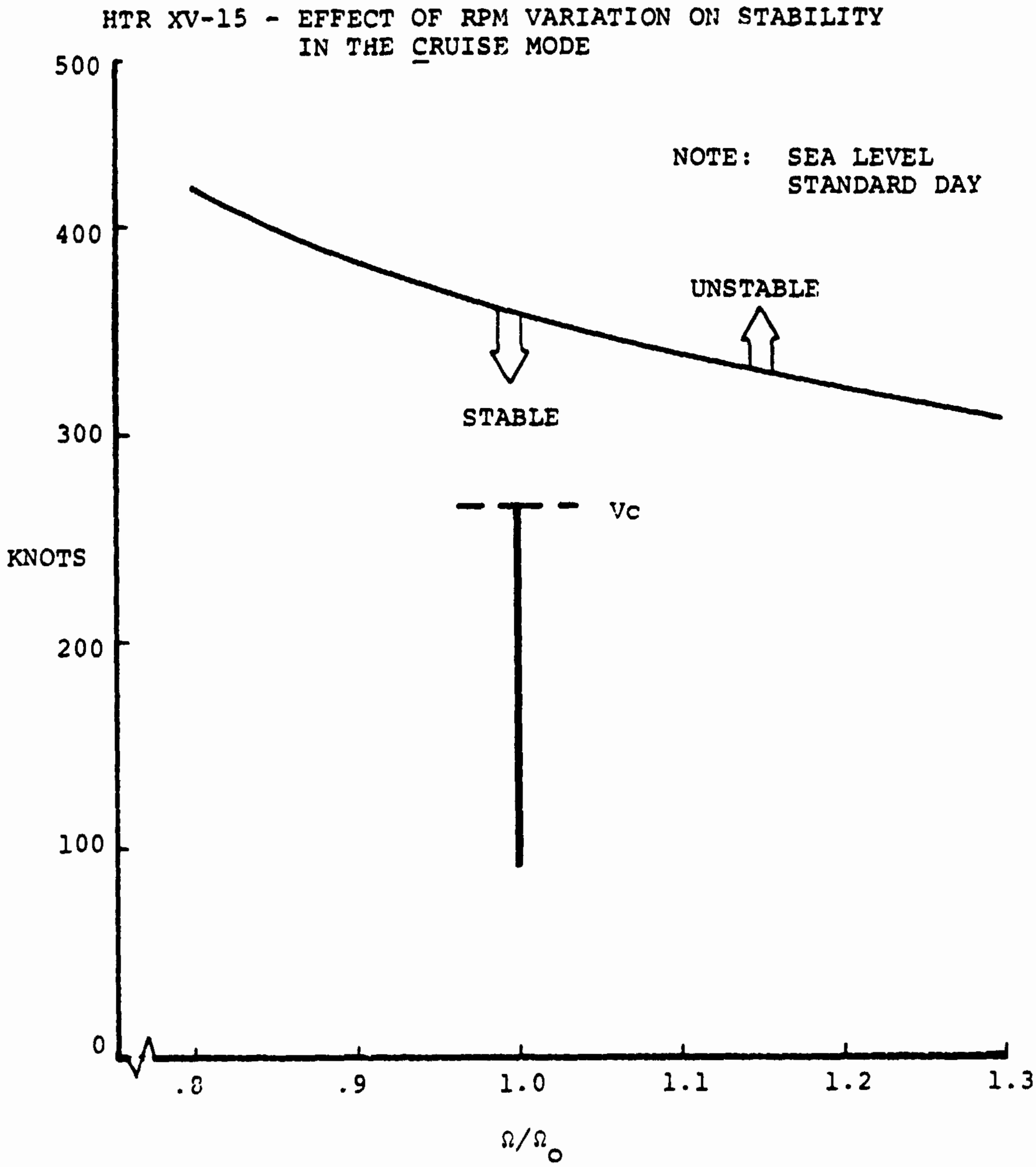


FIGURE 3.4.3. HIGH SPEED CRUISE STABILITY VARIATION WITH RPM

a margin of 36% on Vc or 18% on Vd at sea level.

The boundary shown in Figure 3.4.3 occurs in the symmetric modes. The anti-symmetric mode stability was also examined and remained stable to higher speeds.

Figures 3.4.4 and 3.4.5 are root locus plots for symmetric and anti-symmetric conditions versus speed. These show that in the symmetric case, the root associated with wing chordwise bending goes unstable at speeds above 360 knots: in the anti-symmetric case the root associated with wing vertical bending goes unstable.

3.4.4.2 Effect of Altitude

Cruise speed increases with altitude. The critical speed for flutter onset also increases with the results shown in Figure 3.4.6.

3.4.4.3 Low Speed Cruise and Transition

The stability behavior at 0, 30, 60 and 90-degrees of tilt is shown in Figure 3.4.7 in terms of RPM versus forward speed. The minimum flying speed in the cruise mode is around 95-100 knots. At 95 knots there is a small region of mechanical instability from 600 RPM to 630 RPM, i.e., 60% above the operating RPM. At 100 knots there is no instability although low damping exists in same RPM range which corresponds to the intersection of the wing bending and blade regressive lag frequencies as RPM increases. At 30° tilt the system is stable although a region of low damping persists around 600 RPM where the frequencies of the regressive lag mode and wing

HTR XV-15: HIGH SPEED STABILITY. SYMMETRIC

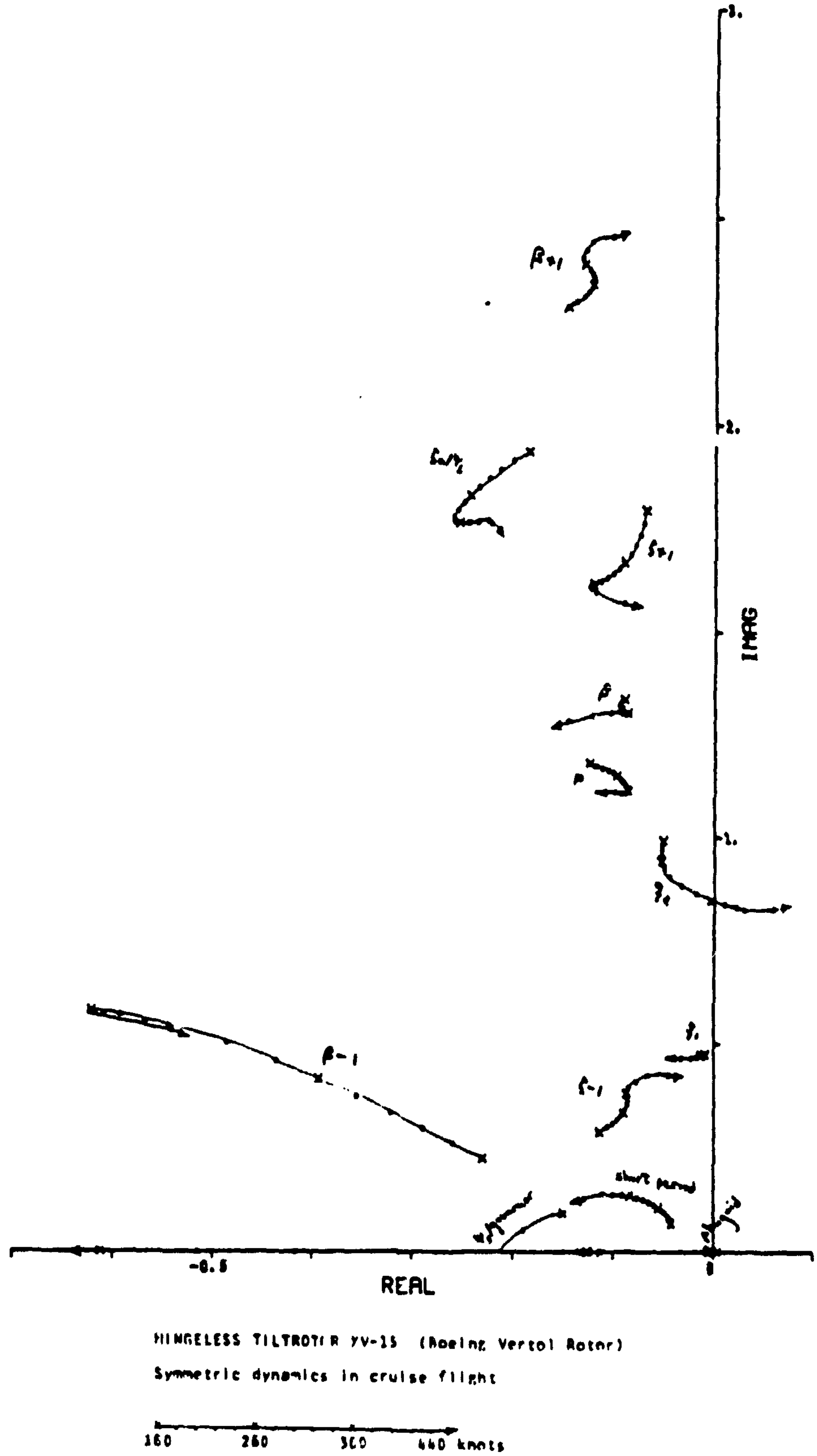


FIGURE 3.4.4. ROOT LOCUS BEHAVIOR OF SYMMETRIC MODES

HTR XV-15: HIGH SPEED CRUISE STABILITY, ANTISYMMETRIC

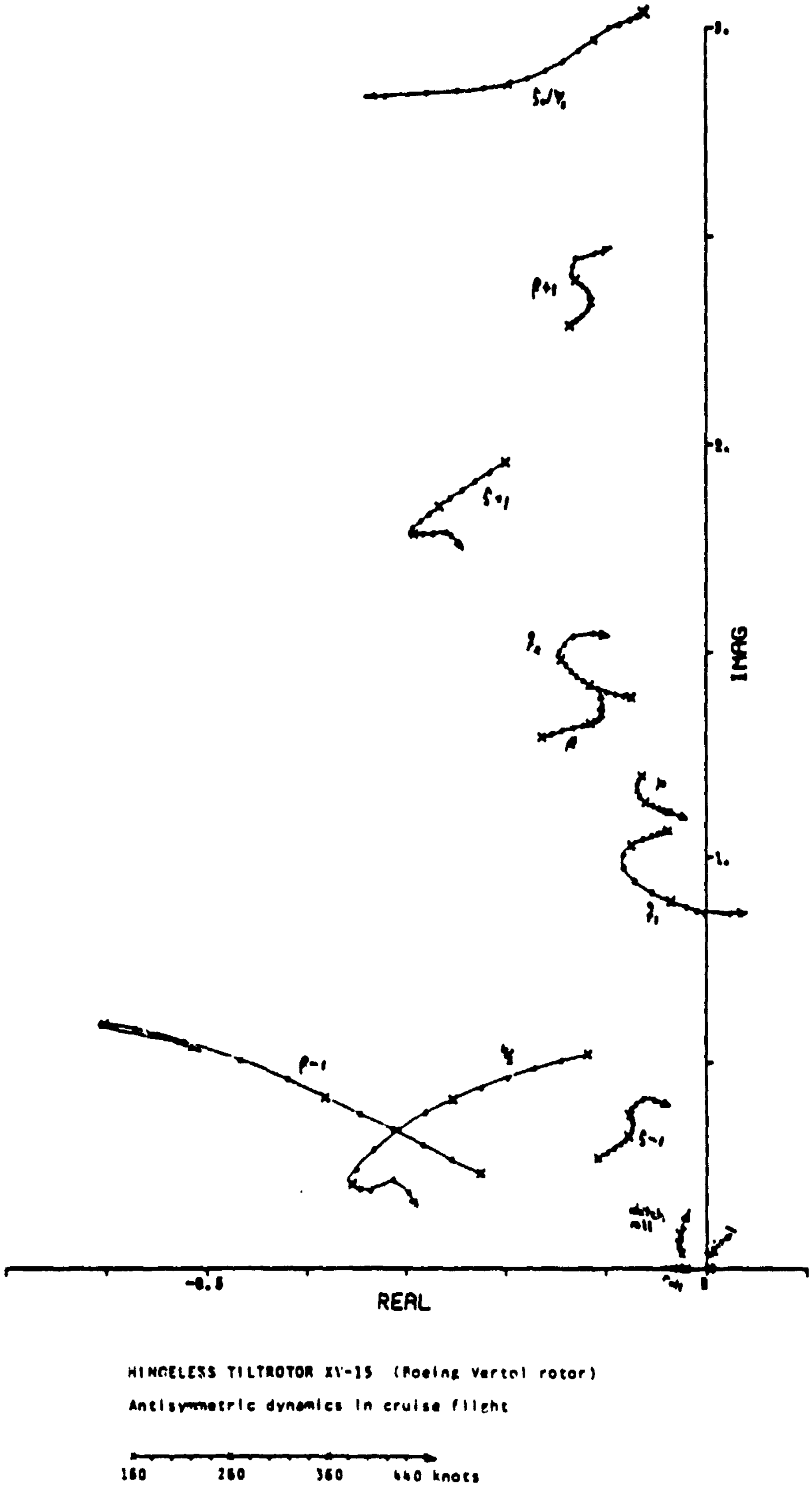


FIGURE 3.4.5 ROOT LOCUS BEHAVIOR OF ANTI-SYMMETRIC MODES

HTR XV-15: CRUISE AND FLUTTER SPEEDS VS ALTITUDE AT 13,586 LBS GROSS WEIGHT

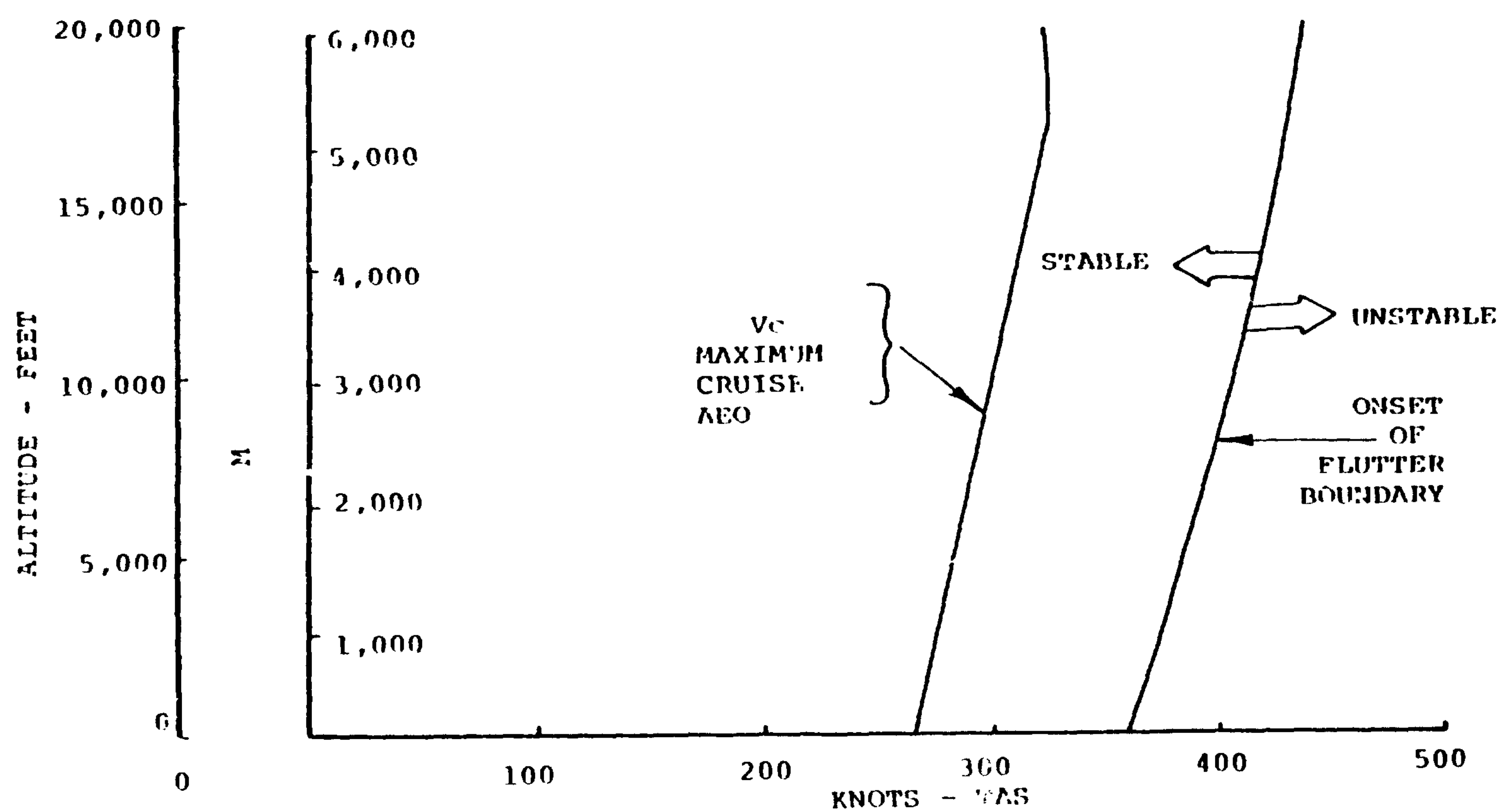


FIGURE 3.4.6 COMPARISON OF FLUTTER SPEED WITH Vc AS FUNCTION OF ALTITUDE

HTR XV-15: STABILITY IN TRANSITION AND LOW SPEED CRUISE

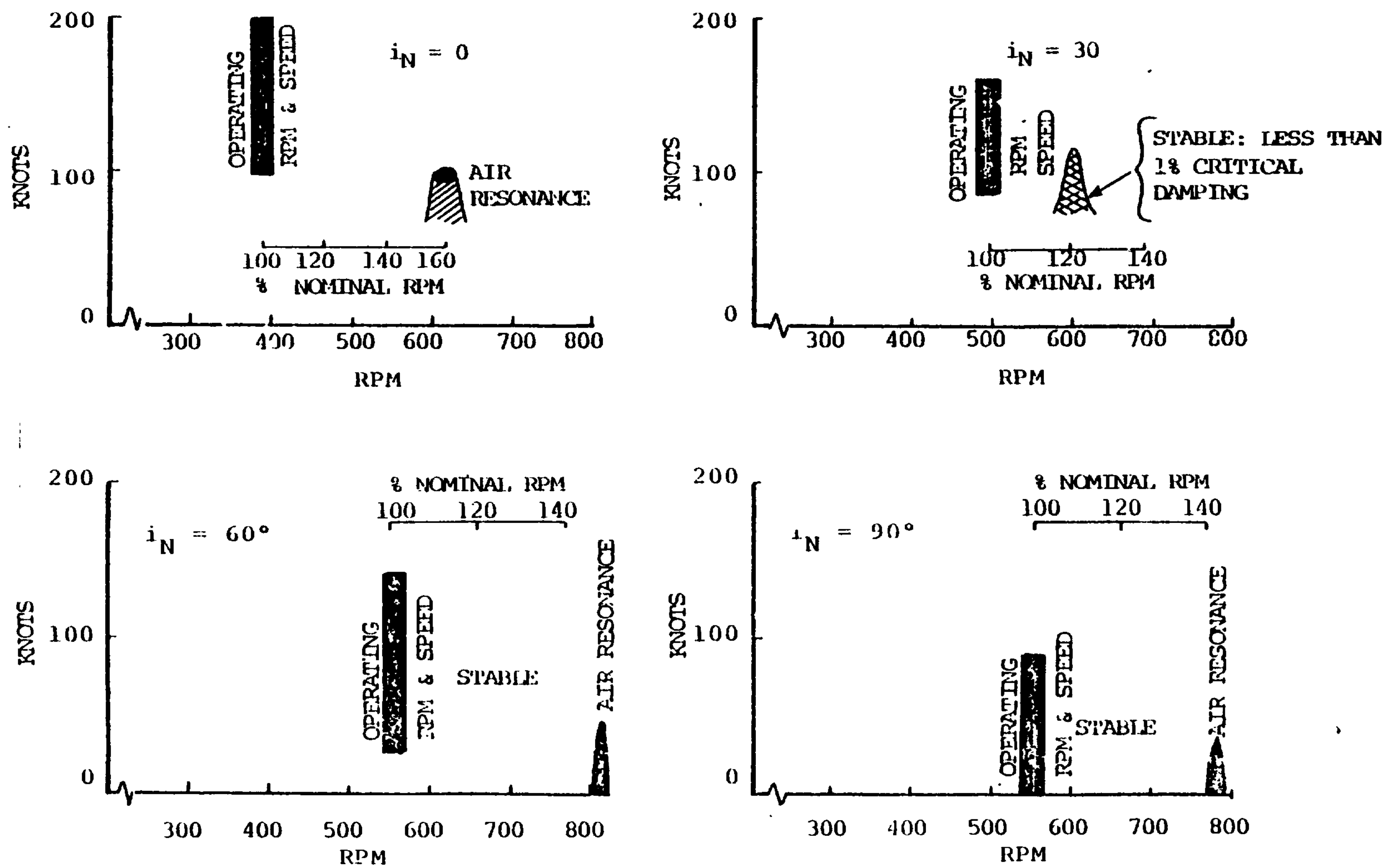


FIGURE 3.4.7. STABILITY THROUGH TRANSITION AND AT LOW SPEED CRUISE

bending coincide.

At 60° tilt there is no longer any significant reduction of damping in this region, but the blade now couples with wing chord bending over a narrow band of RPM slightly in excess of 800 RPM. At 90° tilt this region is still present.

3.4.4.4 Ground Resonance

In the present study, ground resonance was not the subject of detailed investigation. Earlier studies of aircraft of similar gross weight with a rotor having the same dynamic characteristics, indicated that aircraft rigid body frequencies, when supported by the landing gear, were so low that no ground resonance existed. It is expected that a similar situation would be expected in the HTR XV-15. However, to confirm this, detailed information on the XV-15 landing gear would be required and the analysis of ground resonance stability has, therefore, been deferred.

3.4.4.5 Conclusions Regarding Stability

This preliminary evaluation indicated that the aeroelastic and mechanical stability features of the HTR XV-15 are satisfactory. At high speeds the margins are adequate under trimmed flight conditions, and a more detailed study might be necessary to explore accellerated flight or maneuver case if plans to fly the HTR XV-15 become more definite. In transition and at low cruise speed, adequate RPM margins are maintained by scheduling RPM with collective. Overall the aeroelastic stability situation appears to be acceptable.

If the project proceeds to detailed design and fabrication, a more thorough investigation will be required. This will be needed to update the details of the modified wing tip package, particularly the supporting structure of the tilting nacelle.

3.5 STRUCTURAL EVALUATION

3.5.1 Introduction

The structural arrangement for the installation of a Boeing Vertol tilt rotor nacelle on the XV-15 aircraft is shown in Figure 3.5.1. The proposed configuration was selected on the basis of several trade studies which examined the merits of fixed versus tilting engine installations, and geometrical location of the engine as affected by the tilting nacelle and drive system requirements. A brief description of the structure follows.

The tilt nacelle structure is basically a semi-monocoque shell. The rotor loads are transmitted into this shell through the main transmission upper cover attachment bolts. The tilt actuator fitting is attached directly to the nacelle structure. The tilt nacelle is trunnion mounted between two bearing supports which are part of the fixed aft nacelle structure

The aft (fixed) nacelle structure is cantilevered off the XV-15 outer wing by means of a large diameter tapered shaft inserted into the pillow blocks at the two outer rib locations. Rotational fixity is provided by rigidly attaching an extension from the inboard trunnion support area to the wing outboard rib. The powerplant installation is supported at the outboard end of the fixed nacelle structure.

The advantages obtained by this arrangement in terms of reduced complexity in structural build-up and installation of the

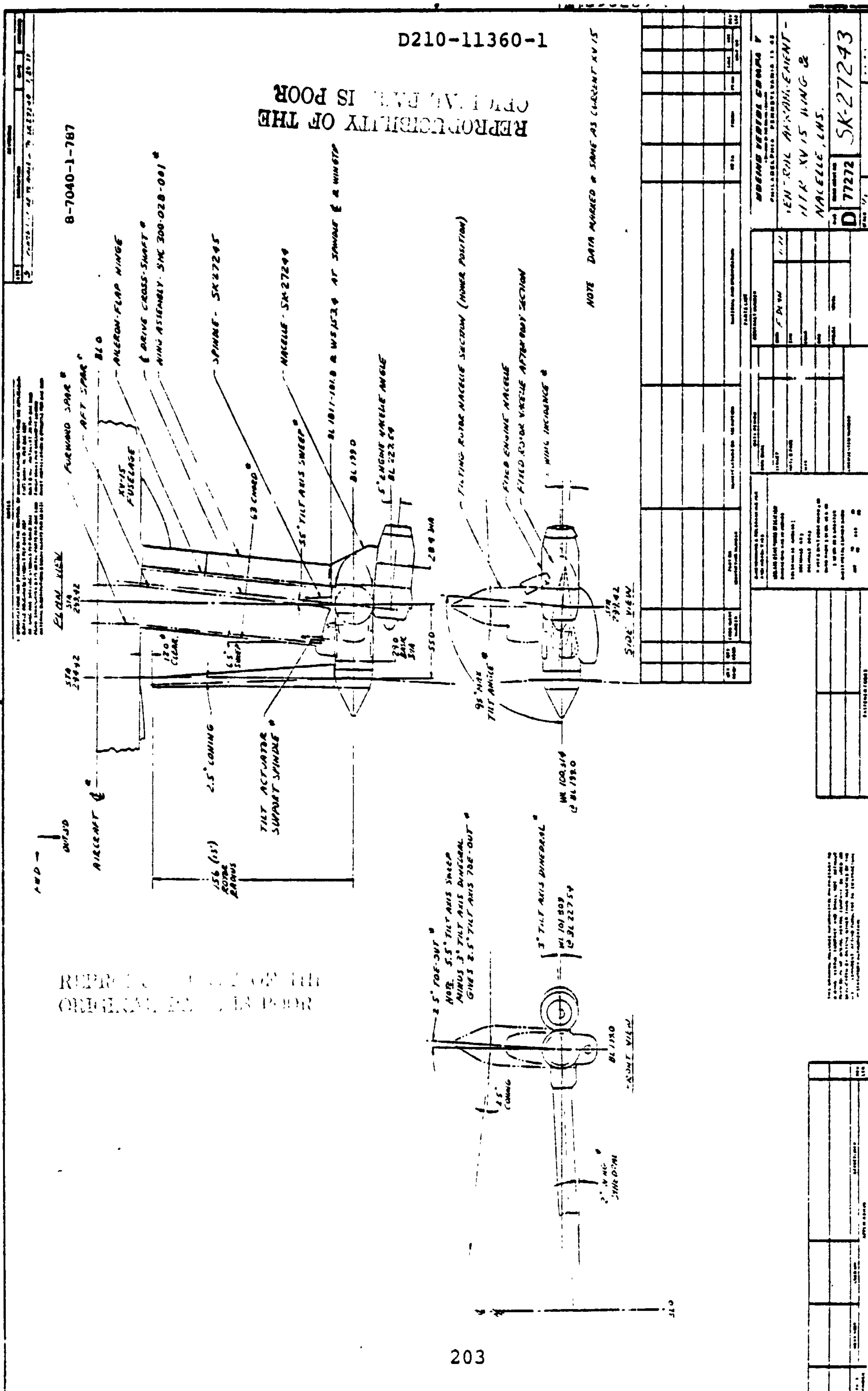


FIGURE 3.5.1. GENERAL ARRANGEMENT HTR XV-15 WING

Boeing Vertol rotor system on XV-15 and the elimination of development requirements for a tilting engine are discussed elsewhere. From the structural point of view, however, the cantilevering of the Boeing Vertol rotor and engine nacelles off the XV-15 pylon support pillow blocks imposes a requirement for the fixed aft nacelle structure to be very stiff in bending and torsion with an attendant weight penalty.

A structural evaluation of the XV-15 with the Boeing Vertol hingeless tilt rotor is given in the following sections. Airframe structural integrity for design flight maneuver induced loadings is investigated first, followed by an evaluation of the structural dynamic characteristics of the installation.

3.5.2 Static Strength

3.5.2.1 Airframe Structural Design Criteria

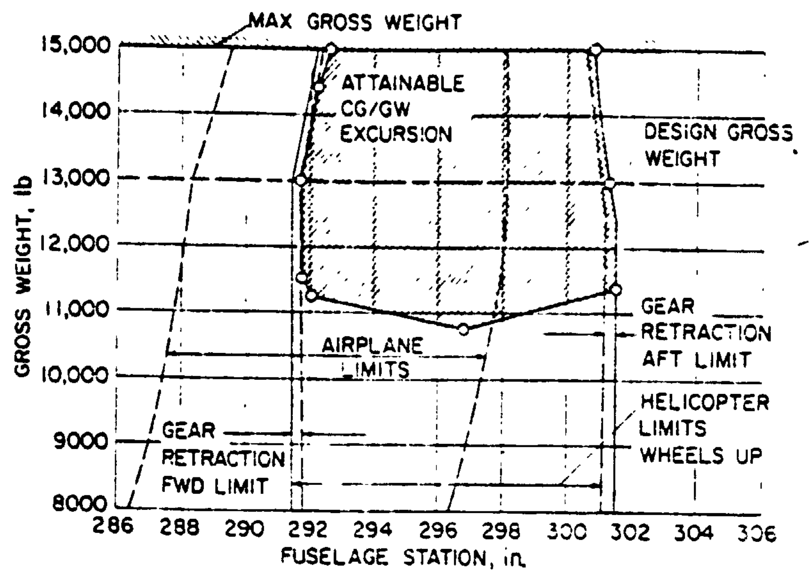
Structural components which are involved in the modifications to install the Boeing Vertol rotor system on the XV-15 will be designed to meet the structural design criteria specified for the XV-15 aircraft. These criteria are summarized below.

a) Aircraft Weight and Center of Gravity.

The aircraft design weights and center of gravity limits are the same as for the XV-15. These are shown in Figure 3.5.2.

b) Design Load Factors - (Flight and Gust Conditions).

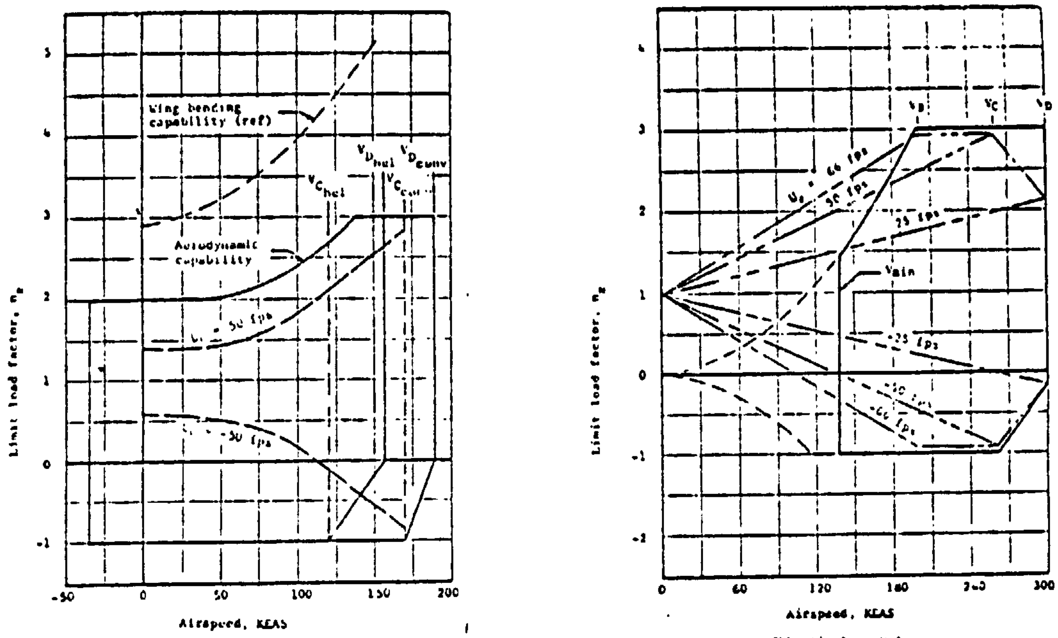
V-n diagrams for the XV-15, shown in Figure 3.5.3 are applicable.



Center of gravity limitations.
HTR XV-15

Condition	Weight (lb)
Design Gross Weight	13,000
Maximum Gross Weight	15,000
Minimum Flying Weight	9,494
Design Landing Weight	13,000
Design Take-off Weight	15,000

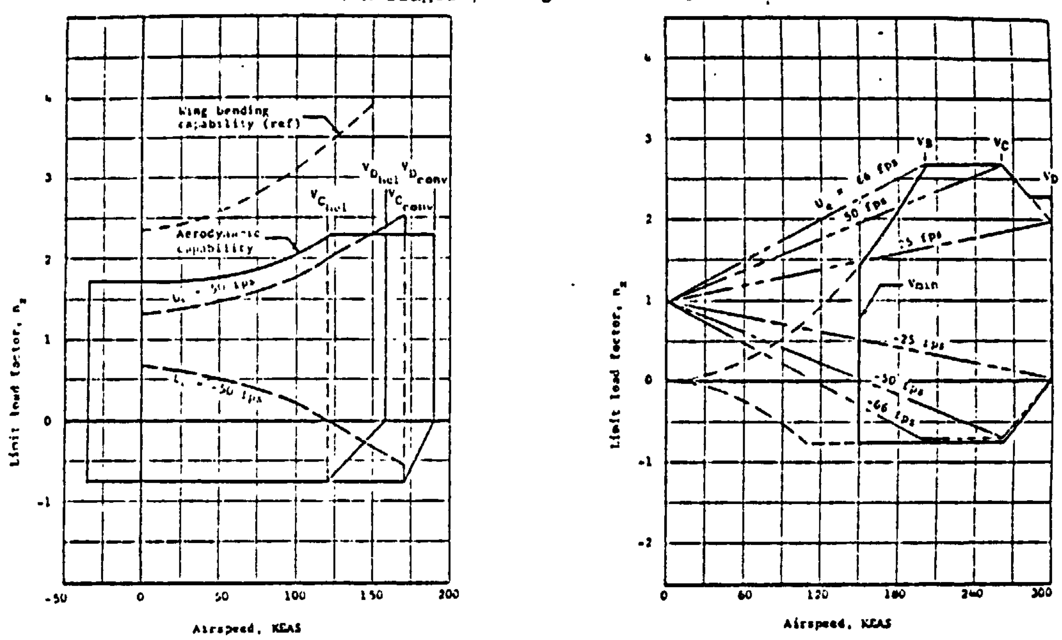
FIGURE 3.5.2. AIRCRAFT WEIGHT AND CENTER OF GRAVITY



(a) Helicopter and Conversion Modes

(b) Airplane Mode

V-n Diagram, Design Gross Weight, 13000 Pounds

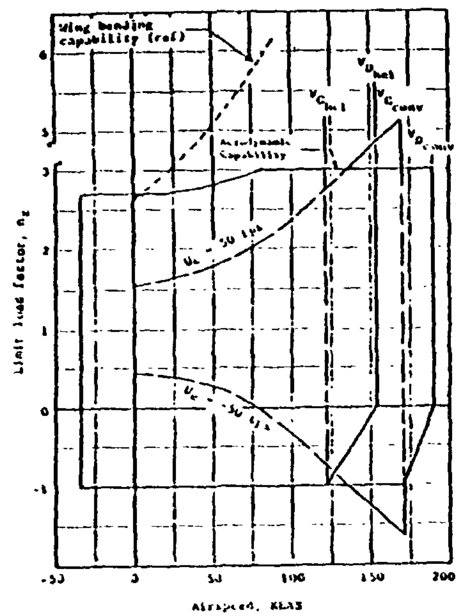


(a) Helicopter and Conversion Modes

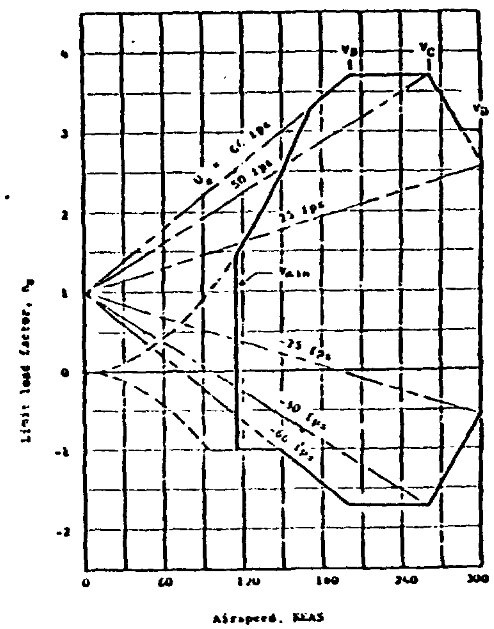
(b) Airplane Mode

V-n Diagram, Maximum Gross Weight, 15000 Pounds

FIGURE 3.5.3. HTR XV-15 DESIGN FLIGHT LOAD FACTORS



(a) Helicopter and Conversion Modes



(b) Airplane Mode

V-n Diagram, Minimum Flying Weight, 9494 Pounds

FIGURE 3.5.3. HTR XV-15 DESIGN FLIGHT LOAD FACTORS (CONTINUED)

c) Flight Maneuver Loading Conditions.

Flight maneuver load conditions are basically the same as for the XV-15 modified to reflect design rotor speeds for the Boeing Vertol 26-foot diameter rotor. The design conditions are set out in Table 3.5.1. Design airspeeds are shown in Table 3.5.2.

d) Ground Loading Conditions.

Landing, ground taxi and handling conditions for the XV-15, shown in Tables 3.5.3 and 3.5.4 are applicable.

e) Miscellaneous Loading Conditions and Safety Factors.

Design crash load factors and safety factors specified for the XV-15 are applicable. These are summarized in Tables 3.5.5.

f) Structural Dynamics.

The flutter and divergence criteria specified for the XV-15 shall not be compromised by the structural modifications involved in Boeing Vertol tilt rotor equipped XV-15 aircraft. Also, these modifications shall not result in increased levels of vibration over that of the current XV-15 aircraft.

g) Service Life.

The design service life for new airframe structural components shall not be less than 5,000 hours. Fatigue life evaluation will be based on the flight spectrums shown in Reference 4.

TABLE 3.5.1. SUMMARY OF FLIGHT LOADING CONDITIONS

Condition [illegible]	Flight Mode	Type of Maneuver	Condition	n_z	Gust Vel. (FPS)	Gust Allev. Factor	A/C Speeds (KT) (1)	Rotor RPM
[illegible]	Helo.	Symm.	Max. Speed	1.0g	(NA)	(NA)	V_D, V_{SIDE}, V_{REAR}	551 & 572
[illegible]	Helo.	Symm.	Dive & Pullout	(Note 2)	(NA)	(NA)	V_A & V_D	551 & 572
[illegible]	Helo.	Symm.	Vertical T.O.	(Note 2)	(NA)	(NA)	0	572
IV	Helo.	Symm.	Push Over	(Note 2)	(NA)	(NA)	V_A & V_D	551 - 572
[illegible]	Helo.	Unsymm.	Rolling Pullout	(See Spec)	(NA)	(NA)	V_D	551 & 572
[illegible]	Helo.	Unsymm.	Yawing	1.0g	(NA)	(NA)	V_A & V_D	572
[illegible]	Helo.	--	Vertical Gust	--	±50	1.0	V_C	551 & 572
[illegible]	Helo.	Autorot.	Pullout-Power Off	(Note 2)	(NA)	(NA)	V_{MRD} & V_D	551 - 572
[illegible]	[illegible]	Autorot.	Yawing-Power Off	1.0g	(NA)	(NA)	V_{MRD} & V_D	551 - 572
[illegible]	Conversion	Symm.	Pullout	(Note 2)	(NA)	(NA)	V_D @ 75° -0°	551 - 572
[illegible]	Conversion	Symm.	Pushover	(Note 2)	(NA)	(NA)	V_D @ 75° -0°	551 - 572
[illegible]	Conversion	Unsymm.	Yawing	1.0g	(NA)	(NA)	V_D @ 75° -0°	572
[illegible]	Airplane	Symm.	Pullup	(Note 2)	(NA)	(NA)	V_{MIN} & V_D	386
[illegible]	Airplane	Symm.	Pushover	(Note 2)	(NA)	(NA)	V_{MIN} & V_C	386
[illegible]	Airplane	Symm.	Max. Pitching	--	(NA)	(NA)	V_{MIN} - V_A	386
[illegible]	Airplane	Symm.	Pitch-Checked Man.	(Note 2)	(NA)	(NA)	V_{MIN} - V_D	386
[illegible]	Airplane	--	Symm. Gust	(Note 2)	(NA)	(NA)	(Note 2)	386
[illegible]	Airplane	--	Unsymm. Gust	(Same as above except for air load assumptions)				
[illegible]	Airplane	Unsymm.	Positive G Roll	(Note 2;6)	(NA)	(NA)	$V_A;V_C;V_D$	386
[illegible]	Airplane	Unsymm.	Min. G Roll	(Note 2;6)	(NA)	(NA)	$V_A;V_C;V_D$	386
[illegible]	Airplane	Unsymm.	Yawing	1.0g	(NA)	(NA)	V_A (Note 7)	386

NOTES: (1) SEE TABLE 3.5.1. (2) SEE APPLICABLE V-n DIAGRAM (SECTION 6) AT MODE, WEIGHT, AND SPEED. (3) TIME OF CONTROL DISPLACEMENT IS 0.3 SEC. UNLESS OTHERWISE SPECIFIED. (4) CONDITIONS TO BE FLOWN AT DESIGN, ALTERNATE, AND MINIMUM FLYING GROSS WEIGHTS, AND AT MAXIMUM APPLICABLE FORWARD AND AFT CG LIMITS. (5) POWER-ON CONDITIONS UNLESS NOTED. (6) SEE FAR XX.349(A) & (B). (7) SEE FAR XX.351.

TABLE 3.5.2. DESIGN AIRSPEEDS

Mode	Speed Item	Symbol	Value (Kts.EAS)*	Notes
Helicopter	Design Operating	V_C	120	95° Tilt Angle (Up from WL)
↑	Design Operating	V_C	140	75° Tilt Angle
	Design Dive	V_D	133	95° Tilt Angle
	Design Dive	V_D	156	75° Tilt Angle
	Design Maneuver	V_A	84	
	Rearward Flight	V_{REAR}	35	True Airspeed
	Sideward Flight	V_{SIDE}	35	True Airspeed
↓	Design Limit Flaps Dn.75°	V_{FL75}	111	
	Min. Rate of Descent	V_{MRD}	70	
Helicopter	Design Landing	V_L	0-80	
Conversion	Design Operating	V_C	140	75° Tilt Angle
↑	Design Operating	V_C	170	0° Tilt Angle
	Design Dive	V_D	156	75° Tilt Angle
	Design Dive	V_D	189	45° to 0° Tilt Angle
	Design Limit Flaps Dn.40°	V_{FL40}	189	
	Stall, Flaps Dn.40°	V_{SF40}	106	
	Min.Operating,Flaps Dn.40°	V_{MIN}	127	$V_{MIN} = 1.2\ V_{SF40}$
↓	Design L.G. Extend/Retract	V_{LE}/V_{LR}	160	
Conversion	Design Landing	V_L	80	
Airplane	Design Operating	V_C	260	0° Tilt Angle, All Cases
↑	Design Dive	V_D	300	
	Design Manouvering	V_A	200	
	Design for Max. Gust	V_B	200	
	Flutter-free Speed	$V_{FLUTTER}$	360	
↓	Stall, Flaps Up	V_{SFO}	115	Design Gross Wt.
Airplane	Min. Operating, Flaps Up	V_{MIN}	138	$V_{MIN} = 1.2\ V_{SFO}$ D.G.W.

* Except as Noted

Condition Number	Type of Landing	Limit Sink Speed (FPS)	Ult. Sink Speed (FPS)	Speed (KN)	L/W Ratio	μ Tire	Reference
XXI	Level, 3-point	10.0	12.25	0-80	1.0	--	FAR XX.479 & (c)
XXII	Level, 2-point	10.0	12.25	0-80	1.0	--	FAR XX.479 & (c)
XXIII(a)	STOL Lat.Drift,2-Pt.	7.5	9.2	80	1.0	0.5	(c)
XXIII(b)	STOL Lat.Drift,3-Pt.	7.5	9.2	80	1.0	0.5	(c)
XXIV	(Main & Nose Gear Obstruction Cases Deleted)						
XXV	Tail Down	10.0	12.25	0-80	1.0	--	FAR XX.481 & (c)
XXVI	One Wheel	10.0	12.25	0-80	1.0	--	FAR XX.483 & (c)

NOTES: (1) Design Landing Weight = 13,000 Lb. (All Cases).

(2) Conditions are at maximum applicable forward and aft C.G. limits.

TABLE 3.5.3. SUMMARY OF LANDING CONDITIONS

TABLE 3.5.4. GROUND TAXI AND HANDLING CONDITIONS

Condition Number	Type of Condition	A/C Weight (LB)	A/C Lift (LB)	Reference
XXVII	Braked Roll - 2-Point	15,000	0	FAR XX.505
XXVIII	Braked Roll - 3-Point	15,000	0	FAR XX.505
XXIX	Nose-Wheel Yaw	15,000	0	FAR XX.509
XXX	Reverse Braking	15,000	0	FAR XX.513
XXXI	Turning	15,000	0	FAR XX.507
XXXII	Pivoting	15,000	0	FAR XX.511
XXXIII	2G Taxi	15,000	0	ANC-2, Para. 3.5
XXXIV	Towing	15,000	0	FAR XX.515
XXXV	Jacking	15,000	0	ANC-2, Para. 4.3
XXXVI	Hoisting	15,000	0	ANC-2, Para. 4.4

NOTE: Conditions are at max. forward and aft c.g. limits.

TABLE 3.5.5. MISCELLANEOUS LOADING CONDITIONS AND SAFETY FACTORS

Factor	Value
Safety, Limit to Yield	1.00
Safety, Limit to Yield (Castings)	1.15
Safety, Limit to Ultimate	1.50
Fitting	1.15
Casting	1.25
Engine Torque	1.25
Other Torque (Drive System, Rotor)	(TBD)
Crash Load, Nacelle Items (Ultimate)*	
n_x (Forward)	7g
n_y (Sideward)	±8g
n_z (Downward, Helo Mode)	15g
n_z (Downward, Conv. Mode)	10g
n_z (Downward, Airplane Mode)	10g
n_z (Upward)	-5g
Crash Load, Wing - Fus. Attach. (Ultimate)**	
n_x (Forward)	15g
n_y (Sideward)	±10g
n_z (Downward, Helo. Mode)	15g
n_z (Downward, Conv. Mode)	10g
n_z (Downward, Airplane Mode)	10g
n_z (Upward)	-5g

* Pertains to nacelle support structure also. Loads factors to act separately at nacelle c.g.

** Acting separately

3.5.2.2 Design Loads

A preliminary review of the flight design conditions shown in Table 3.5.1 indicated that critical structural design loadings will be provided by the following conditions.

Mode	Design Conditions
Helicopter	II, III, V, VI and VII
Conversion	X and XII
Airplane	XVIII (a) and (b)

Design loads for the listed flight conditions were obtained by simulating the appropriate flight condition on the mathematical model for the Boeing Vertol HTR XV-15 aircraft developed under a separate NASA contract and described in Reference 1.

In addition, based upon available structural analysis data for Boeing Model 222 aircraft, Reference 5 and Bell Model 301 (XV-15) Reference 4, an additional condition representating maximum cyclic pitch application to the rotor blades was included.

The results showed that except during gust conditions where the hub moments for the HTR XV-15 aircraft are higher than those used in the Bell design, the modifications need be critically evaluated for the helicopter mode design conditions listed below. Also, the hub moments for the gust cases only affected the design of local attachment structure in the rotor nacelle area and did not compromise the overall structural integrity of the XV-15 wing.

a) VTO

b) Symmetric pitch maneuver (max cyclic)

c) Yawing

Design rotor loads for the above cases are shown in Table 3.5.6.

3.5.2.3 Structural Analysis

A finite element model of the Boeing Vertol HTR XV-15 wing was developed for NASTRAN analysis. As detail drawings for the XV-15 wing were not available the basic data e.g., geometry, section properties, wing/fuselage attachment joints and design reaction system were estimated from data contained in Reference 4. The rotor nacelle, aft support structure and nacelle actuator system were defined on the basis of preliminary design data. The model was constructed such that the rotor nacelle attitude can be varied from hover through conversion to cruise. Figures 3.5.4 and 3.5.5 show the NASTRAN model in hover and cruise attitudes respectively.

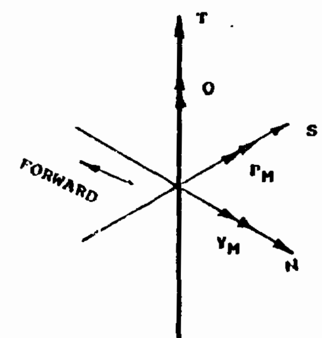
Internal load distribution for the critical helicopter design conditions were obtained from NASTRAN analysis. Figures 3.5.6 and 3.5.7 show the deflected shapes of the HTR XV-15 wing obtained from NASTRAN analysis for two design ultimate loading conditions. The results of this analysis will be used for the detail design of the nacelle and aft support structure. The analytical results shown that the wing root attachment loads are within the structural capability of the XV-15 wing as shown in Reference 4. A few sample calculations follow.

TABLE 3.5.6. HTR-XV-15: ULTIMATE HUB FLIGHT LOADS*
(HELICOPTER MODE)

NO.	FLIGHT CONDITION	C.G.	RPM	RIGHTHAND SIDE ROTOR T	N	S	P	Y	Q	LEFTHAND SIDE ROTOR T	N	S	P	Y	Q
1a	Vertical	F	N_R	20963	893	266	87894	43506	377760	20963	893	-266	87894	-43506	-377760
b	Jump T/O	A			-206	-62	-18743	-10034			-206	62	-18743	10034	
c	at 2g	F	N_{DL}	20963	812	295	90409	48101	343475	20963	812	-295	80409	-48401	-343475
d		A			-188	-68	-20876	-11176			-188	68	-20876	11176	
2a	1g, Hover	F	N_R	10482	1520	330	270252	151869	377760	10482	1520	-330	270252	-151869	-377760
b	7° Aft Cyclic	A													
c	Application	F	N_{DL}	10482	1479	267	319244	182534	343475	10482	1479	-267	319244	-182534	-343475
d		A													
3a	1g, Hover	F	N_R	10482	-1520	-330	-270252	-151869	377760	10482	-1520	330	-270252	151869	-377760
b	7° Fwd Cyclic	A													
c	Application	F	N_{DL}	10482	-1479	-267	-319244	-182524	343475	10482	-1479	267	-319244	182334	-343475
d		A													
4a	1g, Hover	F	N_R	10940	1385	302	246312	135420	377760	10025	-786	171	-139758	78540	-377760
b	Full Pedal	A			1017	221	180758	101580			-1154	251	-205311	115380	
c	Displacement	F	N_{DL}	11067	1317	238	284124	162453	343475	9897	-797	144	-171935	98357	-343475
d	Yaw Left	A			997	180	215078	122975			-1117	202	-240983	137786	
5a	1g, Hover	F	N_R	10025	-786	-171	-139758	-78540	377760	10940	1385	-302	246312	-138420	-377760
b	Full Pedal	A			-1154	-251	-205311	-115380			1017	-221	180758	-101580	
c	Displacement	F	N_{DL}	9897	-797	-144	-171935	-98307	343475	[illegible]	1317	-238	284124	-162457	-343475
d	Yaw Right	A			-1117	-202	-240983	-137786			997	-180	215078	-122975	

NOTES:
1) THRUST INCLUDES 1.075 FACTOR TO INCORPORATE DOWNWASH LOAD ON WING SURFACE.
2) LIMIT TORQUE FACTOR, 1.5.
3) N_R = 551 RPM & N_{DL} = 606 RPM
4) NACELLE AT 90-DEGREES.
5) ALL LOADS SHOWN ARE FOR AIRCRAFT DESIGN GROSS WEIGHT (13,000 LBS)

*THESE ARE APPLIED LOADS AND DO NOT CONTAIN INERTIA RELIEF



SIGN CONVENTION

D210-11360-1

HTR XV-15 - FINITE ELEMENT STRUCTURAL MODEL OF WING AND NACELLE PACKAGE

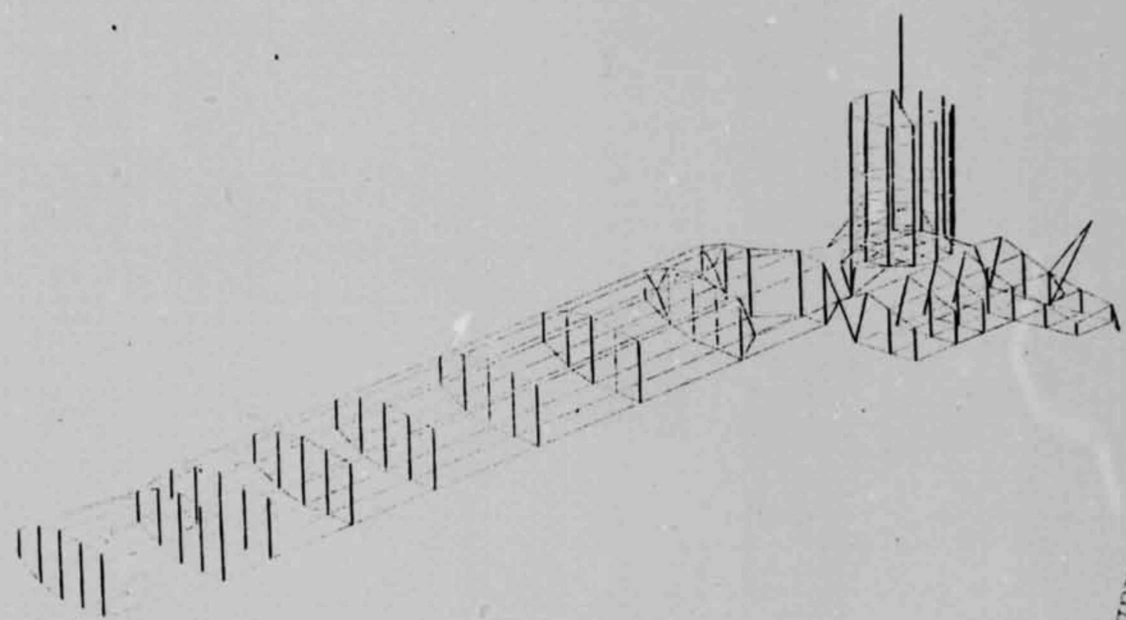


FIGURE 3.5.4. FINITE ELEMENT MODEL - HOVER ATTITUDE

HTR XV-15 - FINITE ELEMENT STRUCTURAL MODEL OF WING AND NACELLE PACKAGE

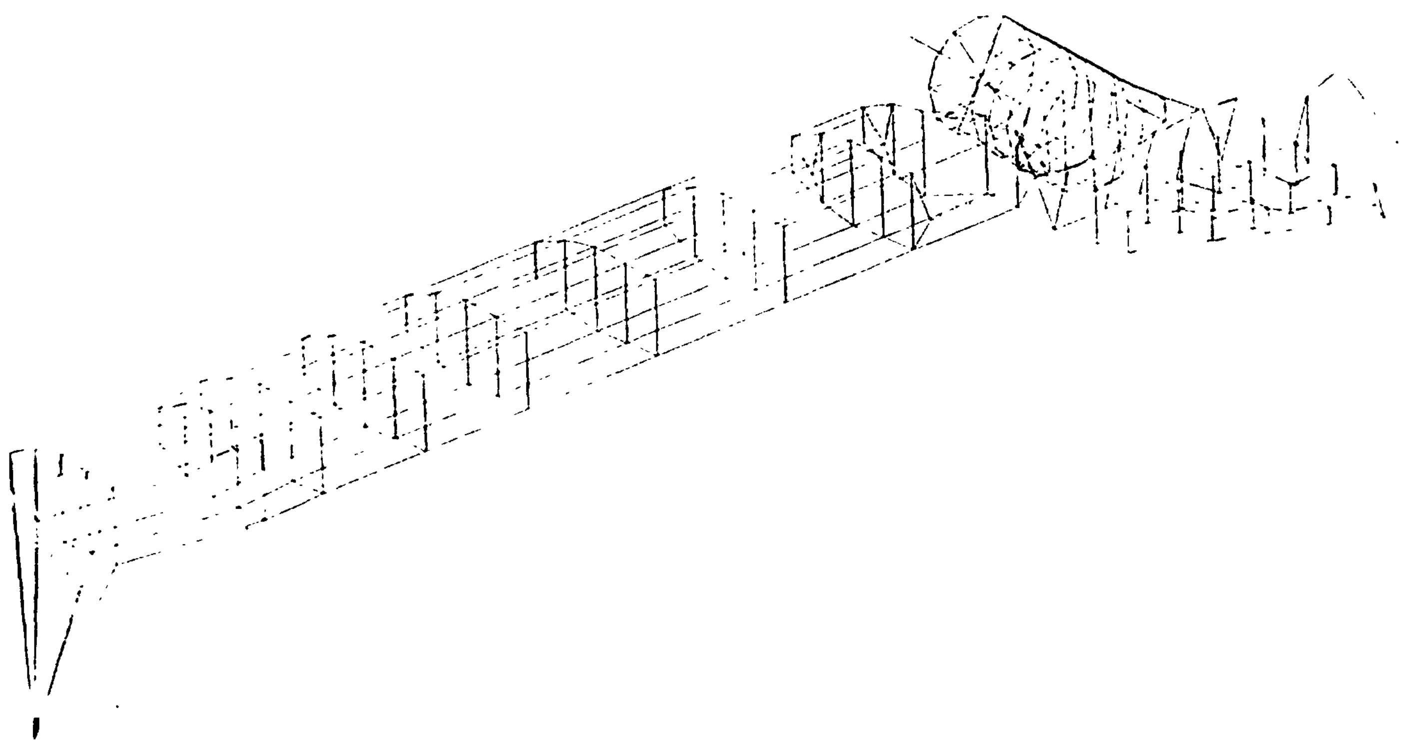


FIGURE 3.5.5. FINITE ELEMENT MODEL - CRUISE ATTITUDE

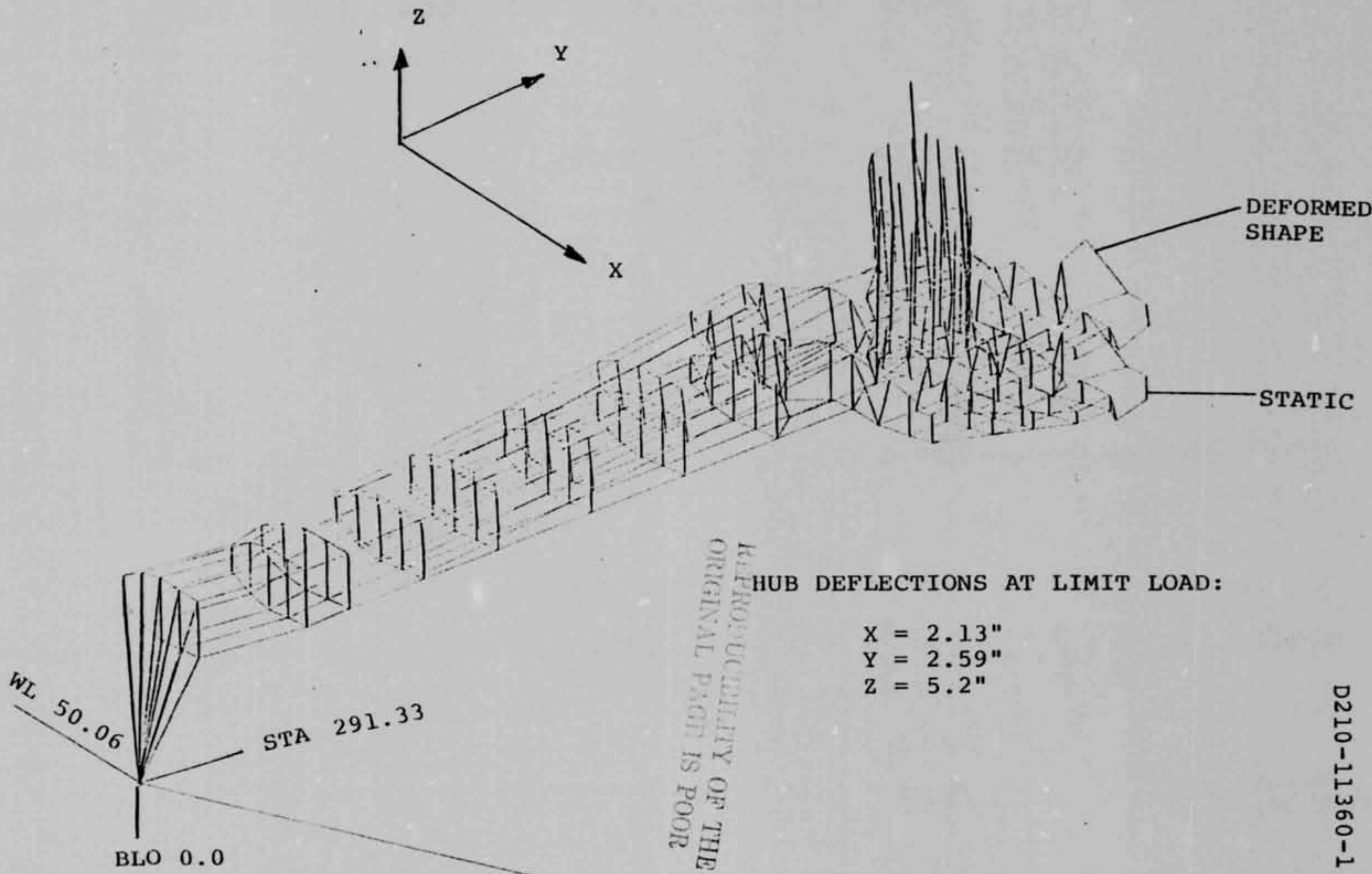


FIGURE 3.5.6. DESIGN CONDITION 1, 2g VTO, DEFLECTIONS

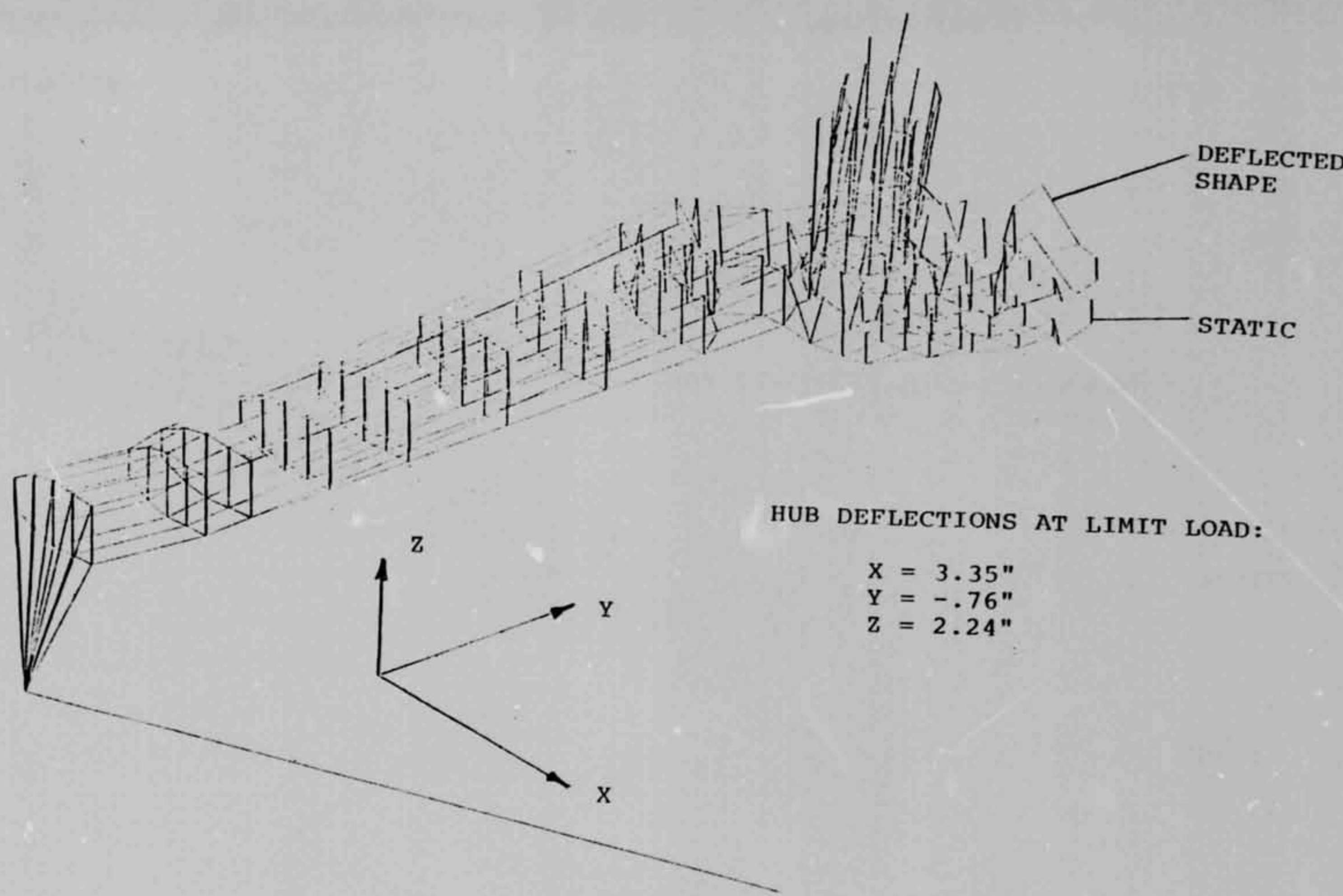


FIGURE 3.5.7. DESIGN CONDITION 3 - APPLICATION OF 7° AFT CYCLIC - MAXIMUM DEFLECTIONS

- Wing/Fuselage Interface

1) Spanwise Bending Moment

HTR XV-15 ultimate bending moment (2g jump takeoff)

$$= 2.39 \times 10^6 \text{ In. Lb.}$$

Dynamic amplification factor = 1.13 (Reference 4)

$\therefore$ Design ultimate bending moment

$$= 2.39 \times 10^6 \times 1.13$$

$$= 2.70 \times 10^6 \text{ In. Lb.}$$

Allowable moment at wing root (from Reference 4)

Limit bending moment = 1.49×10^6 In. Lb.

Ultimate factor = 1.5

Dynamic amplification factor = 1.13

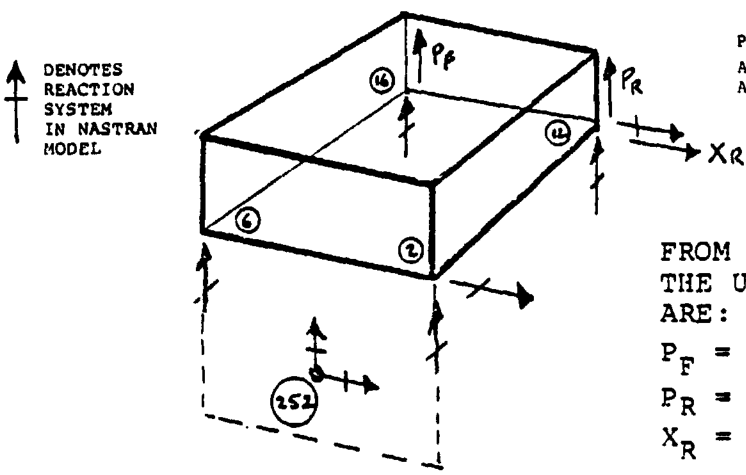
$$\therefore \text{B.M.} = 1.49 \times 10^6 \times 1.5 \times 1.13 \times 1.20$$

$$= 3.031 \times 10^6$$

$$\text{M.S.} = \frac{3.031 \times 10^6}{2.70 \times 10^6} - 1 = 0.12$$

2) Attachment Loads

Critical condition = 2g VTO



P_F, P_R, X_X ARE LOADS APPLIED TO FUSELAGE AT ATTACHMENT POINTS

FROM NASTRAN RESULTS THE ULTIMATE LOADS ARE:

P_F = 9,843 Lbs

P_R = 5,470 LBS

X_R = 635 LBS

These loads are within 5% of the attachment loads for corresponding cases in Reference 4, Table 6-3.

3) Inboard Wing Panel

Max shear flow occurs in the forward upper panel for the 2g VTO condition

$$q_{max} = 727 \text{ Lb/In.}$$

$$\delta_{sn} = \frac{727}{0.160} = 4{,}544 \text{ psi}$$

$$F_{sn} = 42{,}000 \text{ psi (Reference 4)}$$

$$R_S = .108$$

Panel compression:

Combine loads from elements 136 and 142 through 145

$$P = -[(0.5 \times 28928) + 30674 + 43358 + 40104 + (0.5 \times 59832)$$

$$= -158{,}516 \text{ Lbs}$$

Panel Width = 28 Inches

$P_c/\text{in} = 5{,}661$ Lbs

$$f_c = \frac{5661}{2 \times .080} = 35{,}380 \text{ psi}$$

$$F_c = 60{,}400 \text{ psi (Reference 4)}$$

$$R_c = .586$$

Dynamic amplification factor = 1.13

$$\text{M.S.}\ \frac{1}{1.13[R_s^2 + R_c^2]^{1/2}} -1 = 0.485$$

Nacelle Tilt Actuator

NASTRAN element 304

Maximum compression load = 28,886 Lb.

By Reference 4, , Section 6.3.6.2

Allowable load = 21,059 Lb.

Hence actuator requires beef-up

Assuming same proportions, the required effective diameters of the rod are calculated below.

$$\text{Load Ratio } R = \frac{28886}{21059} = 1.372$$

$$I_1' = .121$$

$$I_2' = .440$$

For I_1'

$$\text{I.D.} = 1.030 \text{ In.}$$

$$\text{O.D.} = \left(\frac{64}{\text{J1}} \times I_1' + 1.03^4\right)^{1/4}$$

$$= 1.3765 \text{ In.}$$

For I_2' I.D. = 1.719 In.

$$\text{O.D.} = \left(\frac{64}{\text{J1}} \times .440 + 1.719^4\right)^{1/4}$$

$$= 2.051 \text{ In.}$$

Comparing these with the dimensions shown in the reference, it is seen that they represent a 6% increase in the maximum diameter of the actuator screw rod.

3.5.3 Vibration Characteristics

3.5.3.1 Normal Modes

In order to determine analytically the vibration characteristics of the HTR XV-15 aircraft, the NASTRAN model developed for the static strength analysis of the wing was modified to include a stick representation of the fuselage and empennage. The initial model mass distribution corresponded to the HTR XV-15 AMP weight statement. The free-free normal modes of vibration of the model obtained during preliminary analysis indicated presence of undesirable local engine and nacelle modes. These modes were, to a large extent, suppressed by redefining and stiffening the fixed aft nacelle support structure, beefing-up the rotor nacelle structure bending stiffness and the local stiffness of the rotor nacelle trunnion fittings. The wing symmetric and antisymmetric mode natural frequencies for the HTR XV-15 airframe less rotor blades, obtained from NASTRAN analysis of the model are shown in Table 3.5.7. The table also includes corresponding data obtained from Reference 4, Figures 5-8 and 5-9 for purposes of comparison. The modal displacements for each of the vibratory modes are presented in Figure 3.5.8. The input data for NASTRAN analysis of one case is included as an example in Appendix IV.

TABLE 3.5.7. WIND NATURAL FREQUENCIES (CYCLES/SECOND)

CONFIGURATION	HTR/XV-15		XV-15 (1)	
CONVERSION ANGLE (DEGREES)	90	0 (Cruise)	90	0 (cruise)
1. SYMMETRIC MODES				
1st Beam	2.75	2.827	3.2	3.5
1st Chord	4.22	4.06	5.1	5.6
1st Torsion	7.06	6.45	7.0	6.9
Pylon Yaw	10.39	10.12	10.9	11.7
2. ANTI-SYMMETRIC MODES				
1st Beam	8.58	9.19	8.8	8.9
1st Chord	--	--	4.6	4.8
1st Torsion	6.12	5.84	6.1	5.6
Pylon Yaw	11.55	10.92	13	11.5
(Pylon Pitch)	(7.5)	(6.65)	--	--

(1) Reference 4 , Figures 5-8 and 5-9.

MODE: SYMMETRIC HOVER

FREQUENCY: 2.752

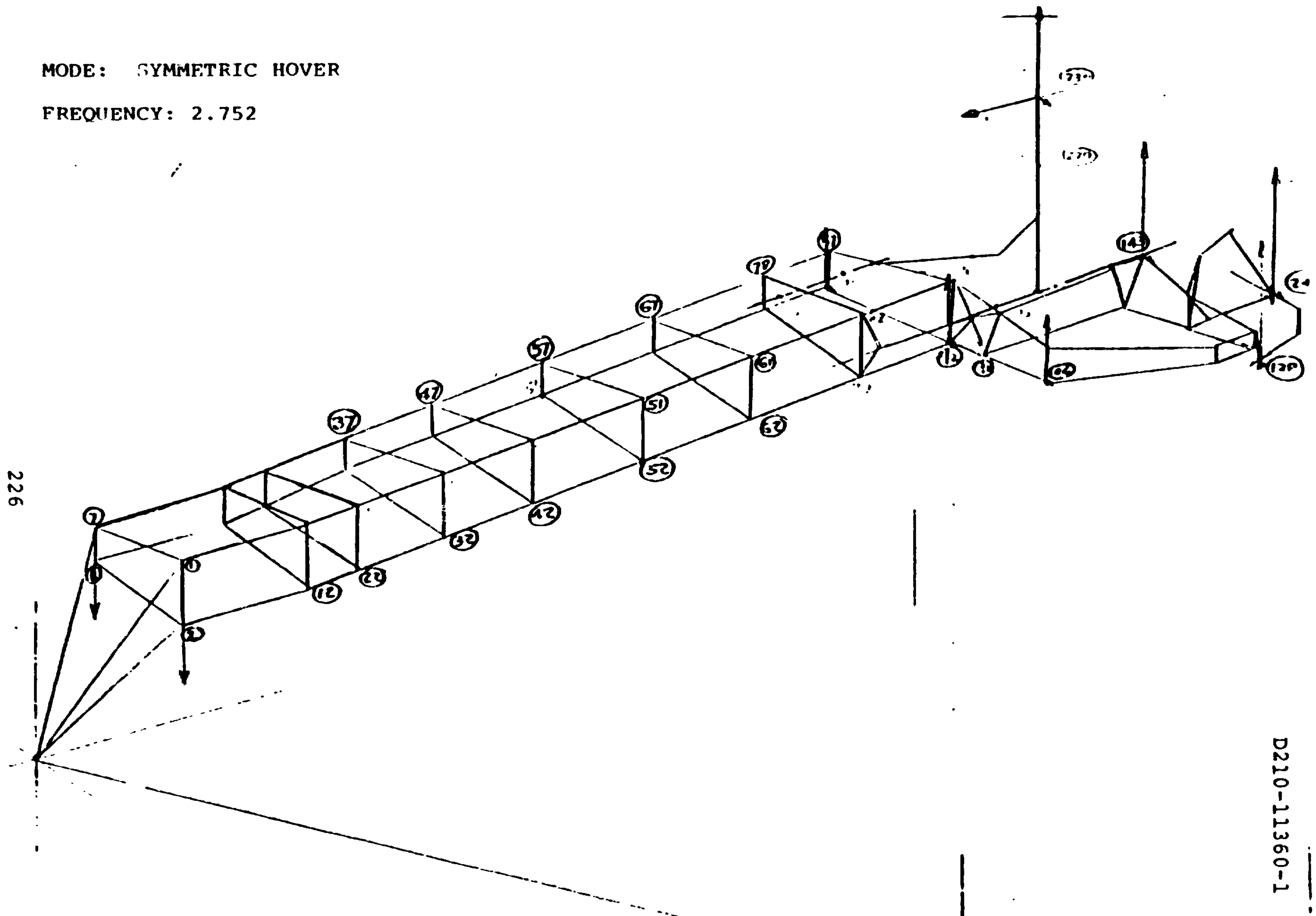


FIGURE 3.5.8. MODEL DISPLACEMENTS

MODE: SYMMETRIC HOVER

FREQUENCY: 4.215

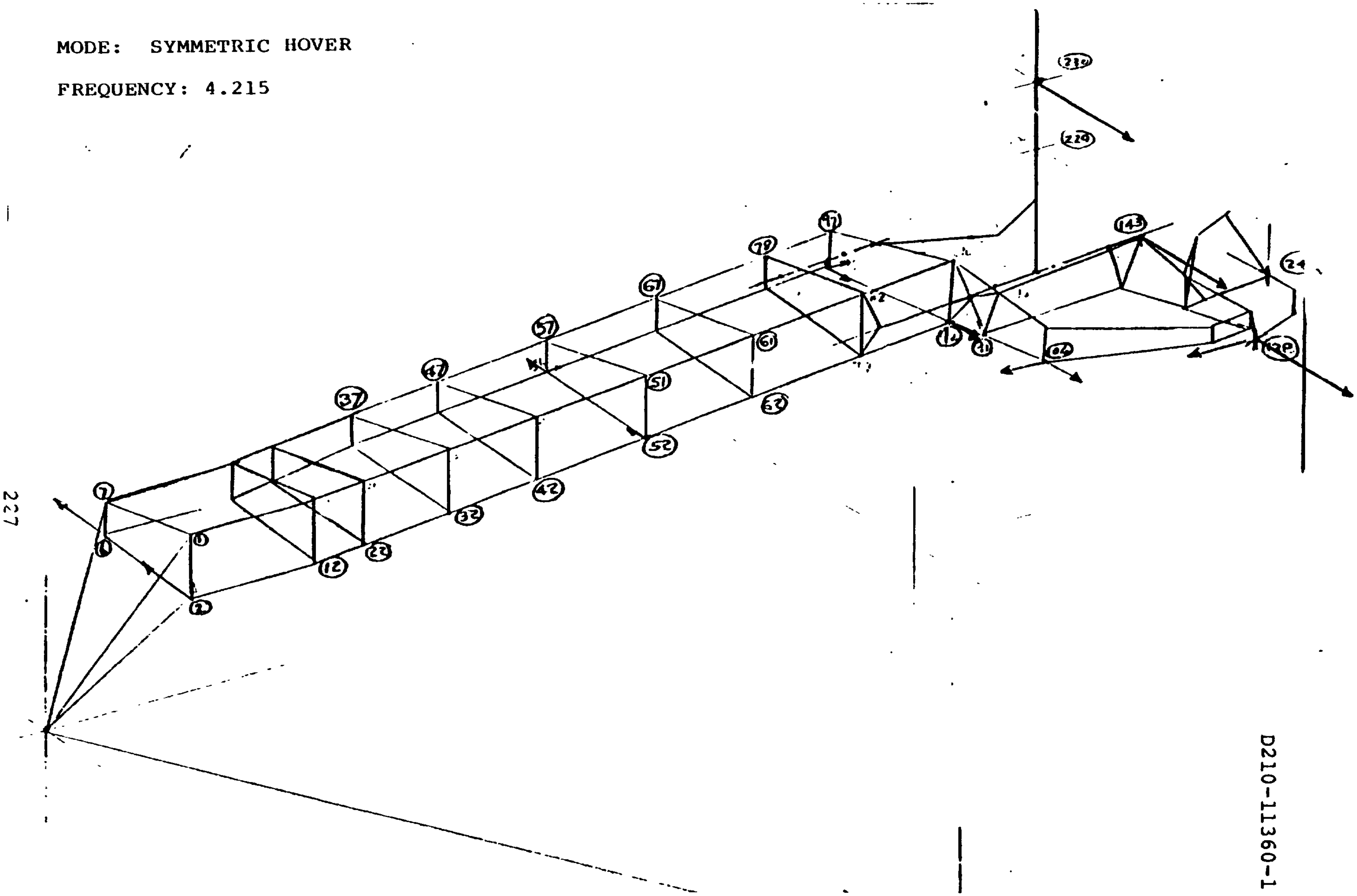


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: SYMMETRIC HOVER

FREQUENCY: 7.060

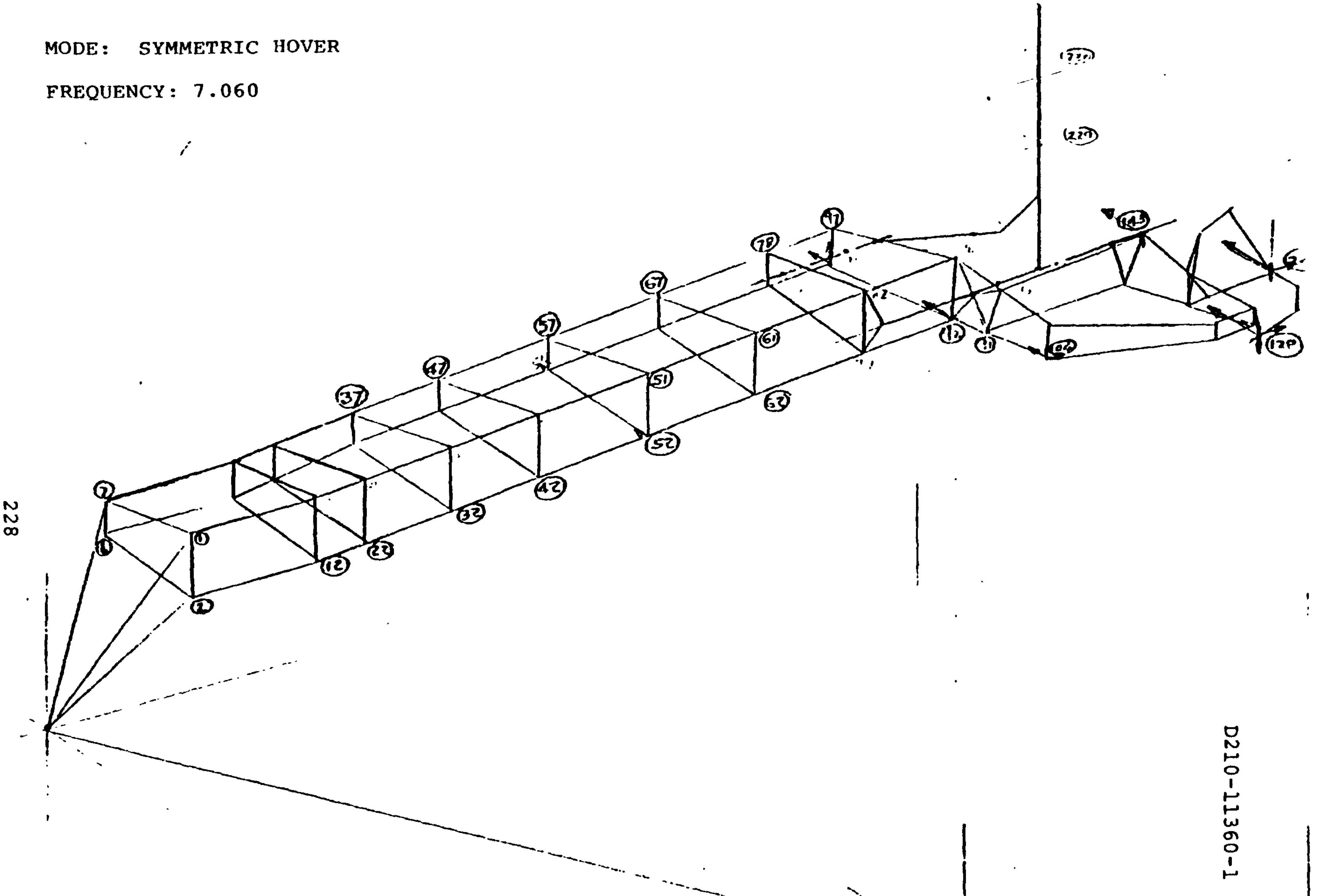


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: SYMMETRIC HOVER

FREQUENCY: 10.39

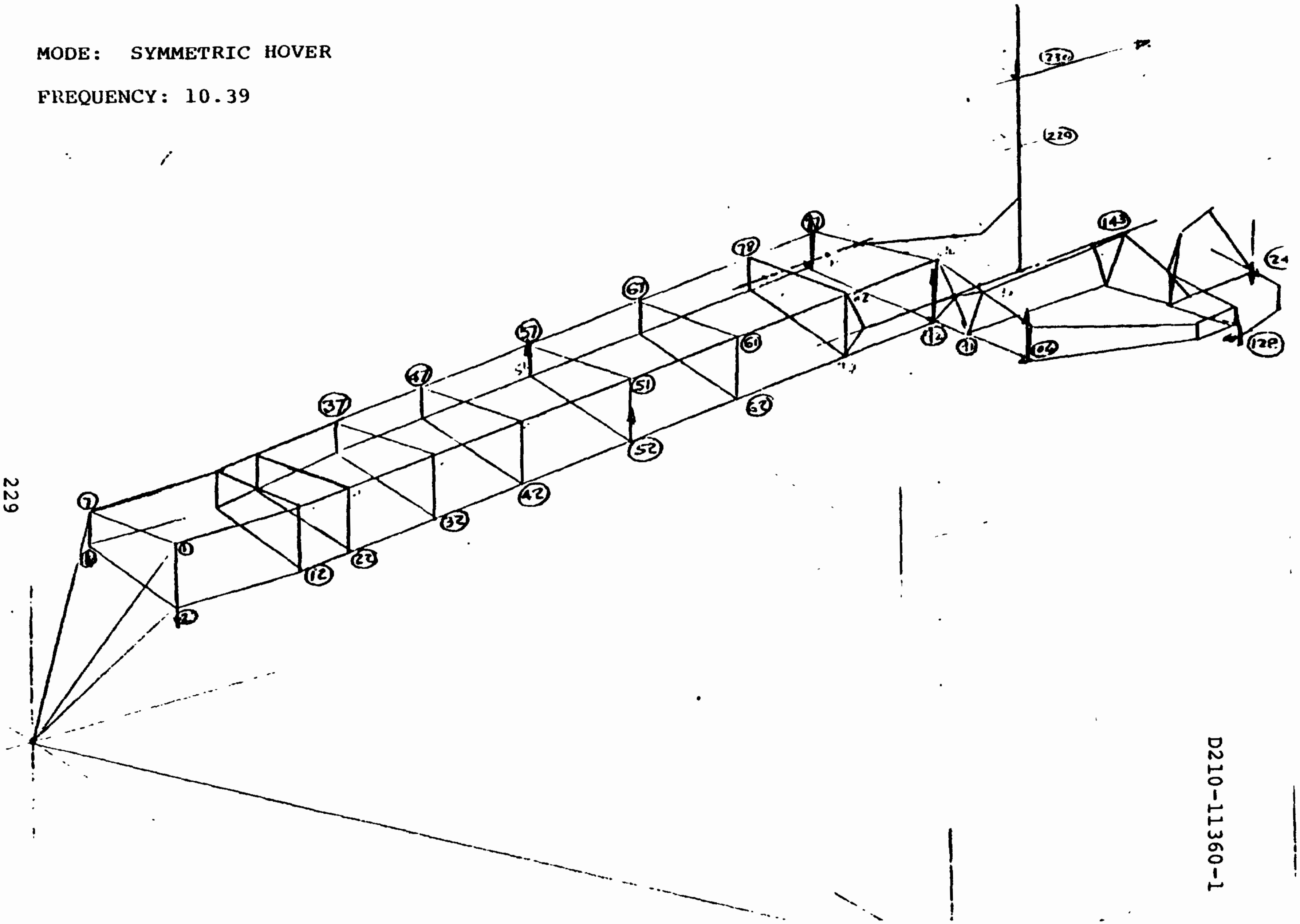


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: SYMMETRIC CRUISE

FREQUENCY: 2.827

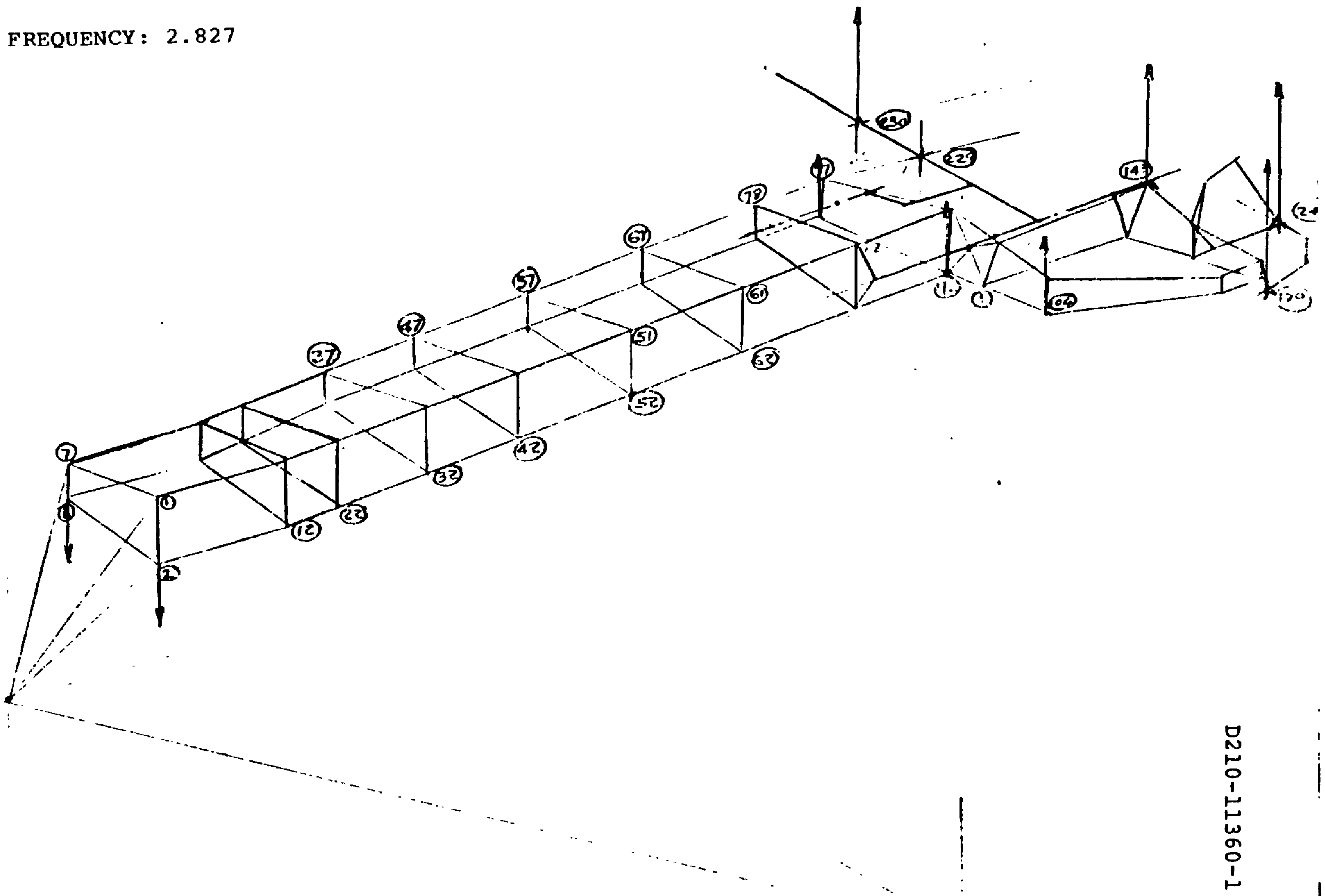


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: SYMMETRIC CRUISE

FREQUENCY 4.055

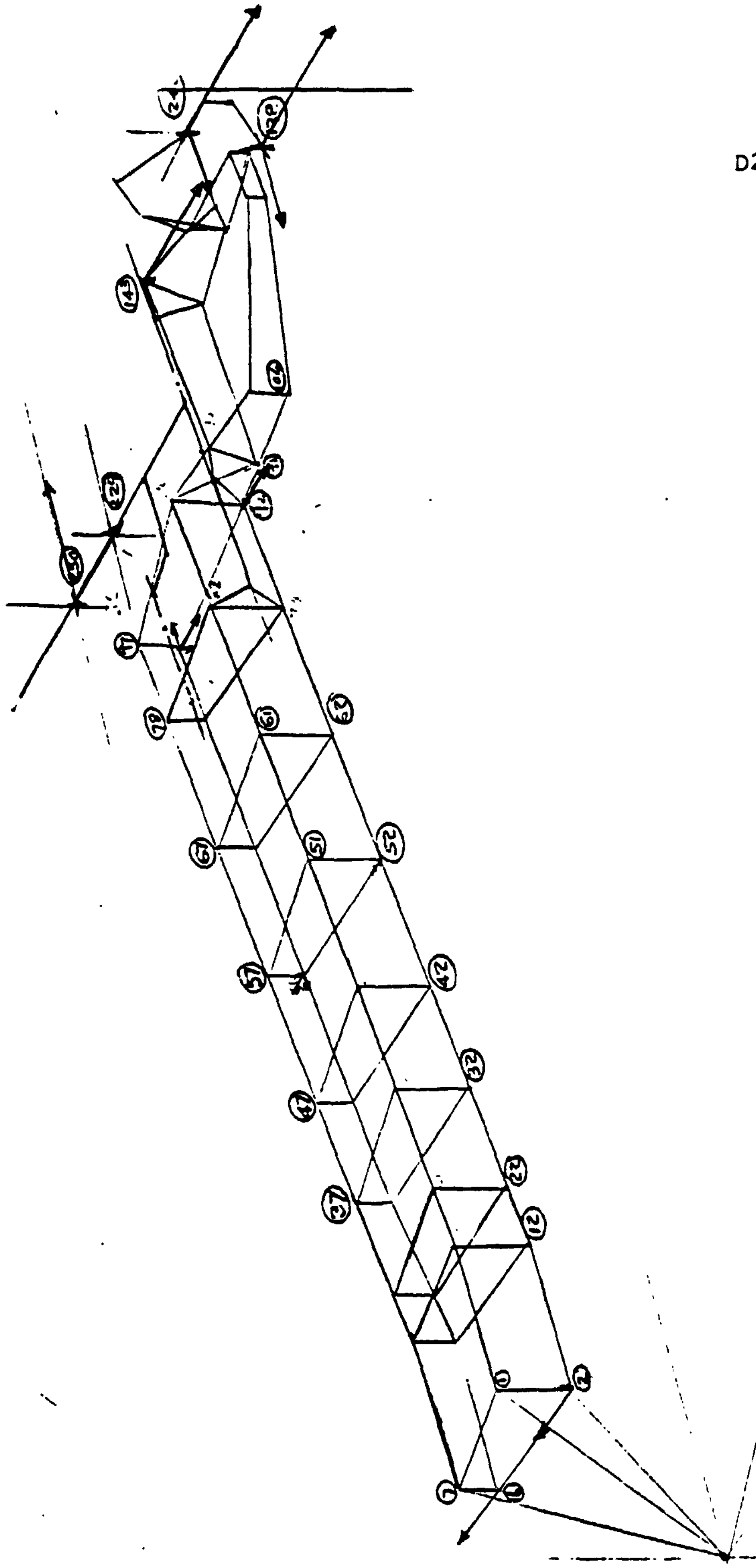


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: SYMMETRIC CRUISE

FREQUENCY: 6.447

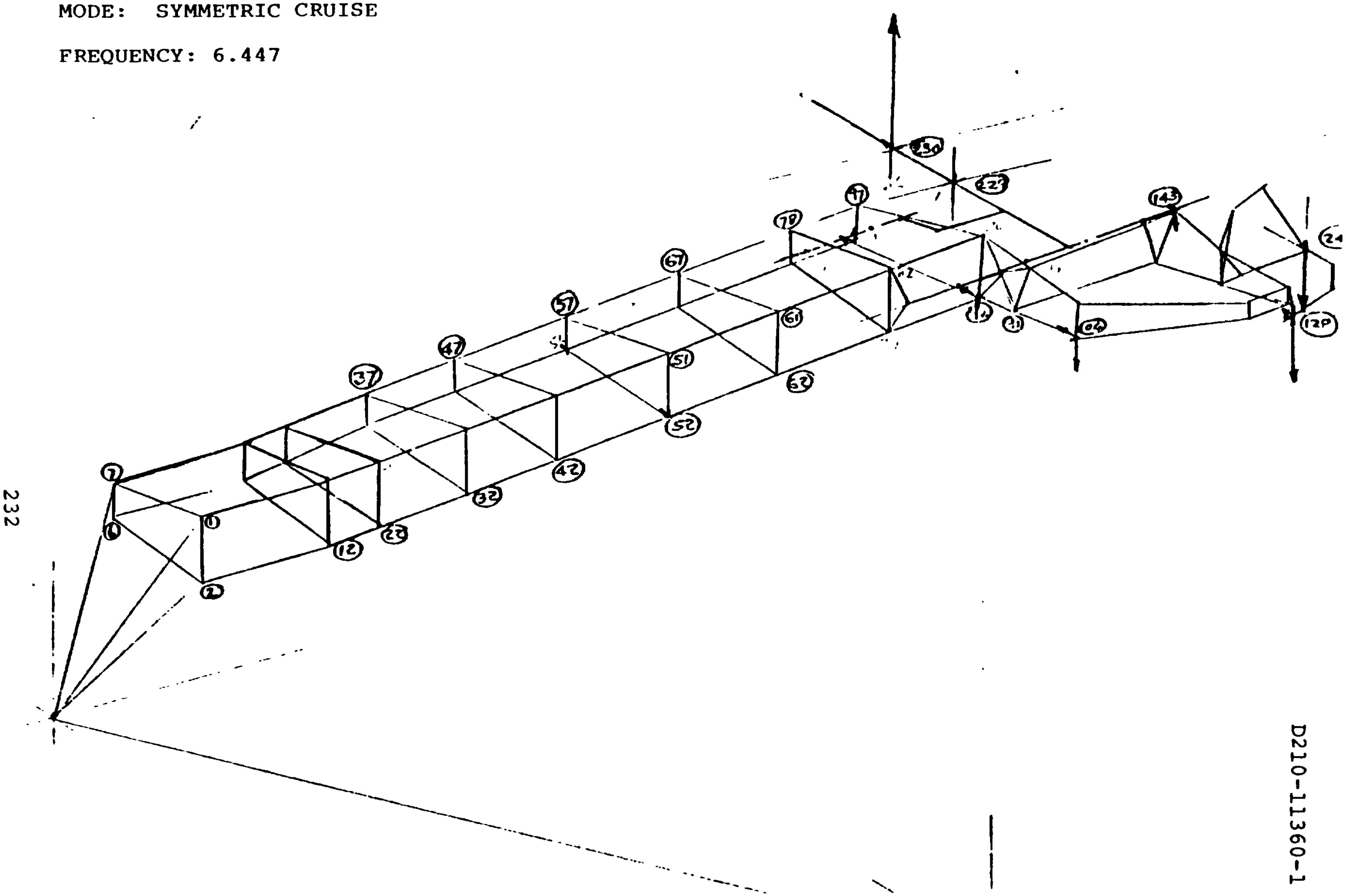


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: SYMMETRIC CRUISE

FREQUENCY: 10.115

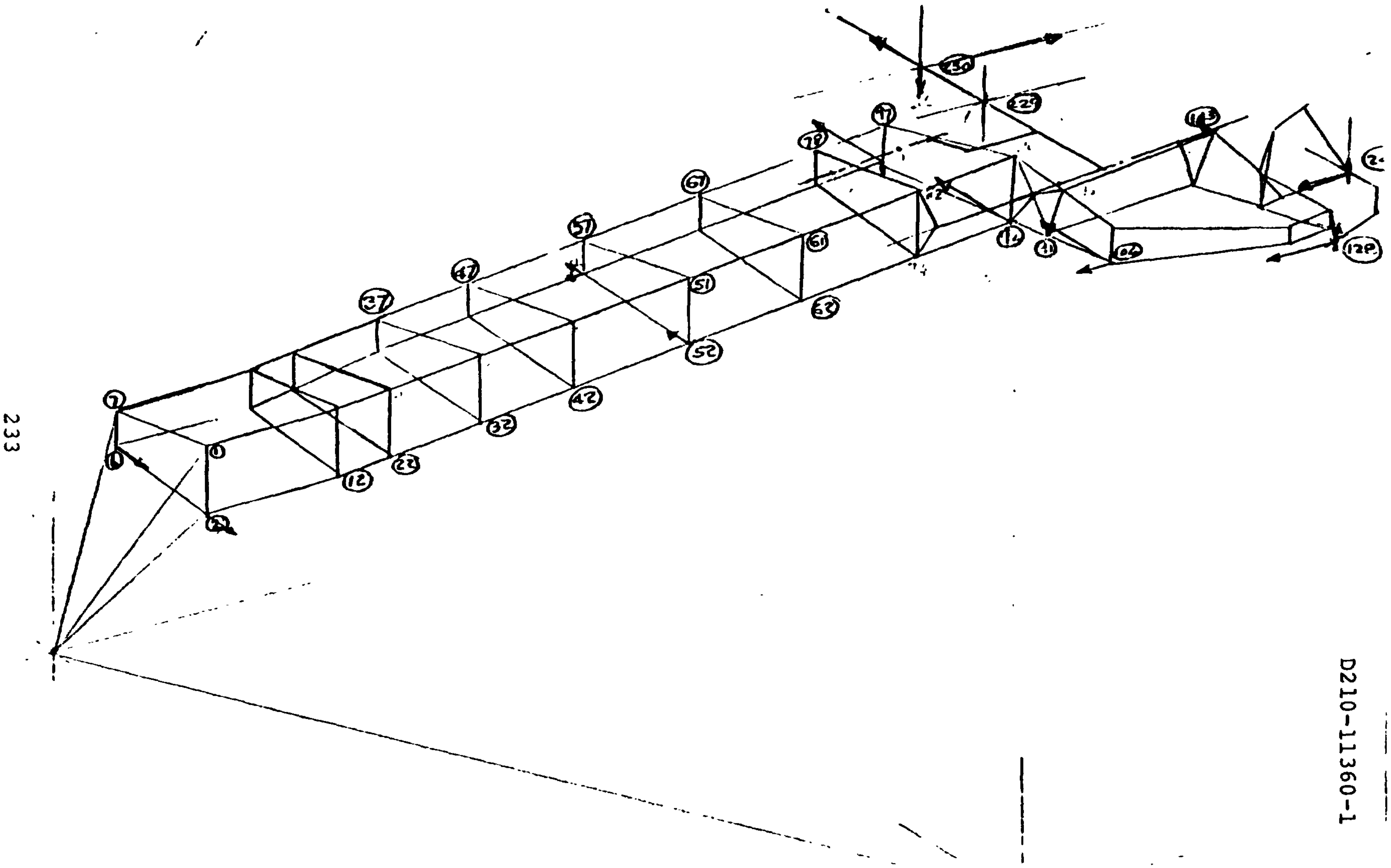


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC HOVER

FREQUENCY: 6.118

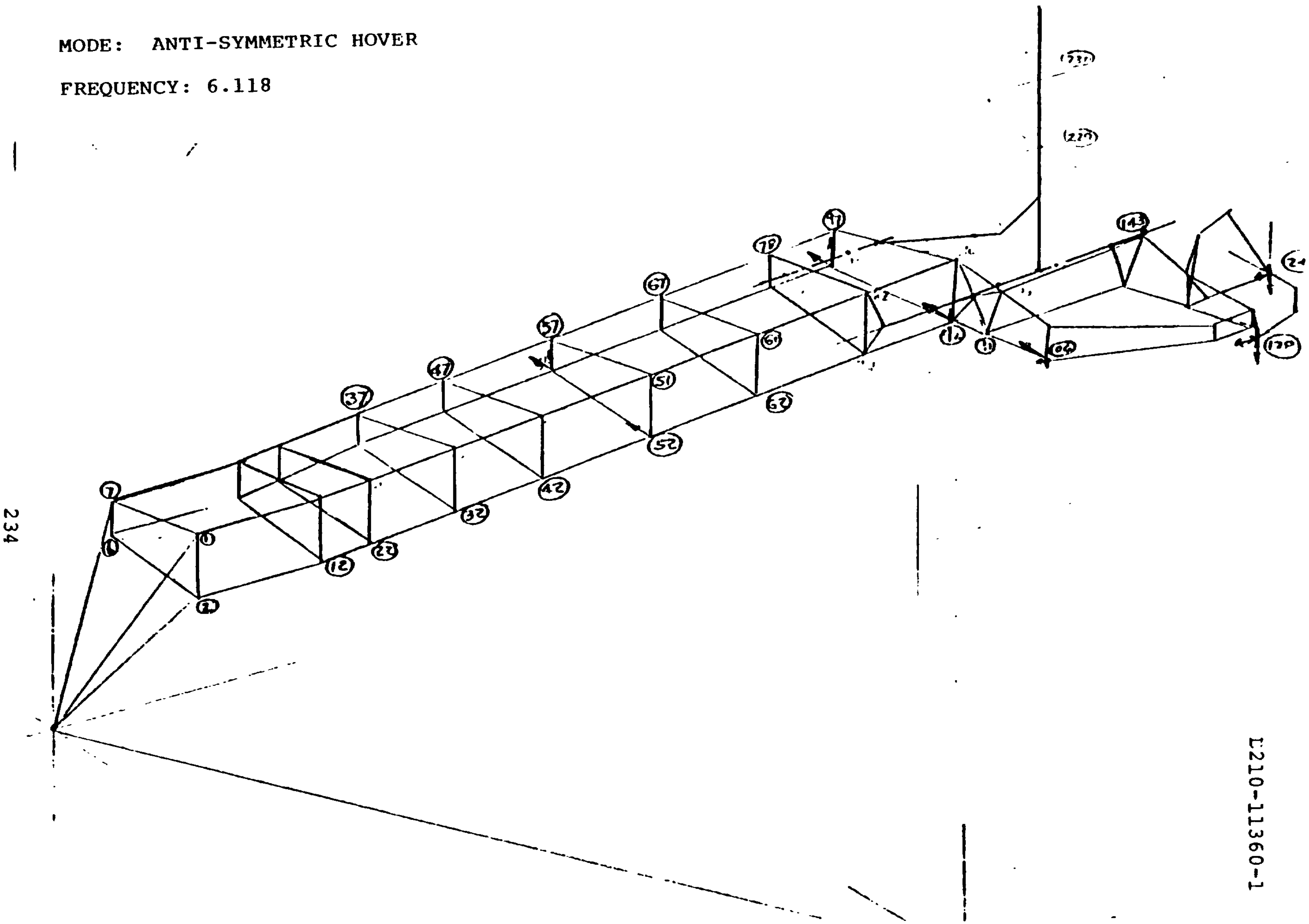


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC HOVER

FREQUENCY: 8.525

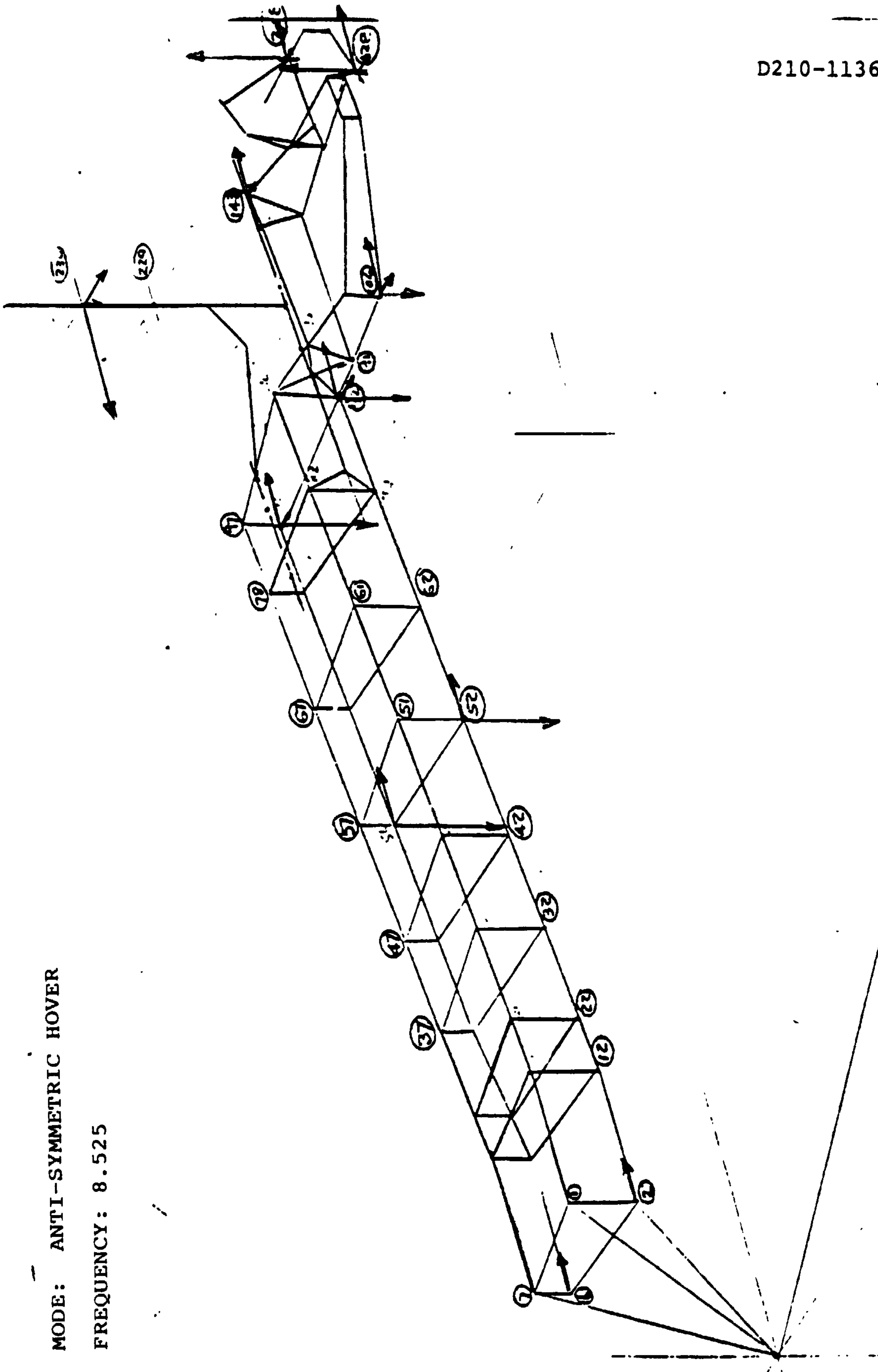


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC HOVER

FREQUENCY: 7.496

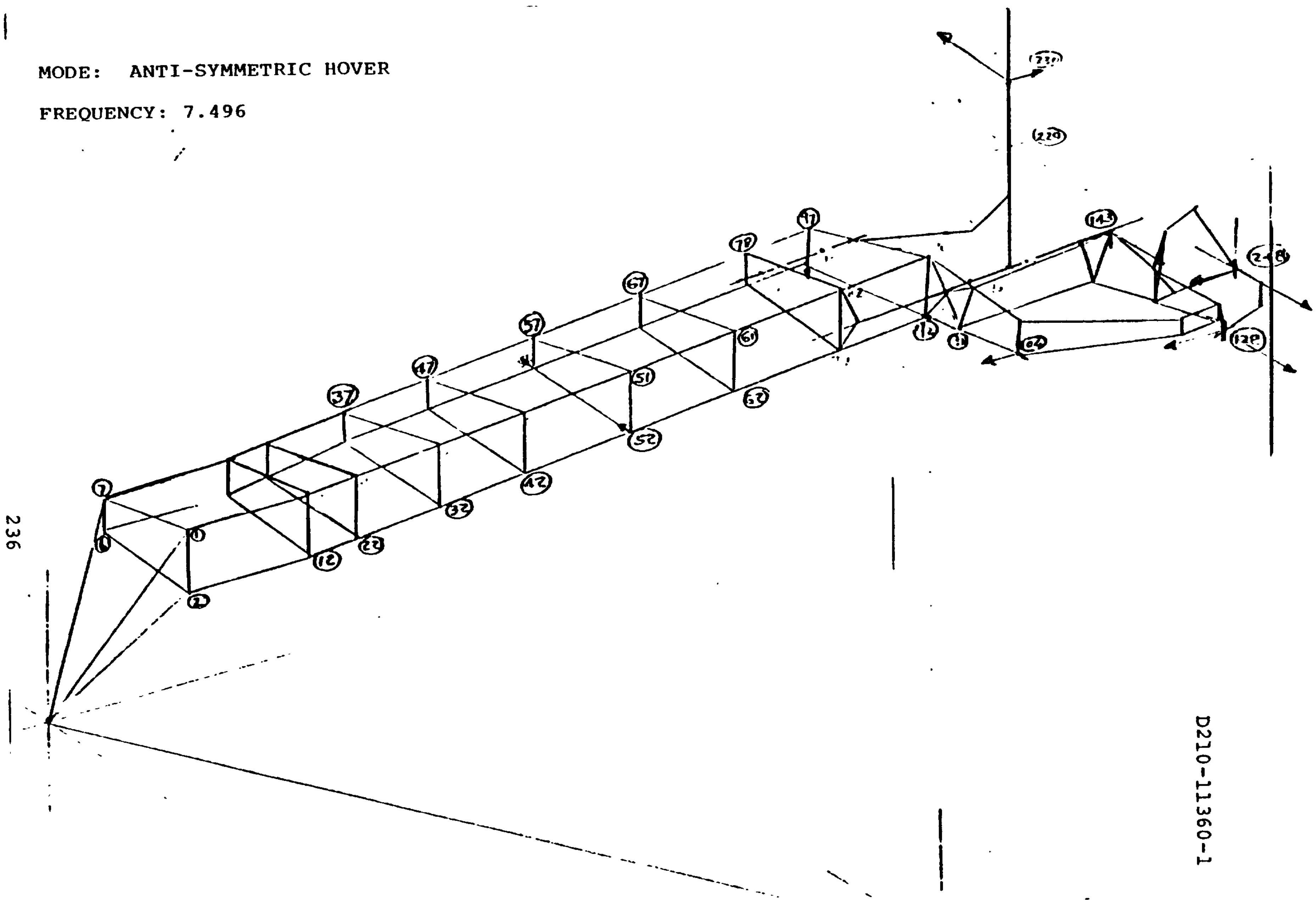


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC HOVER

FREQUENCY: 11.545

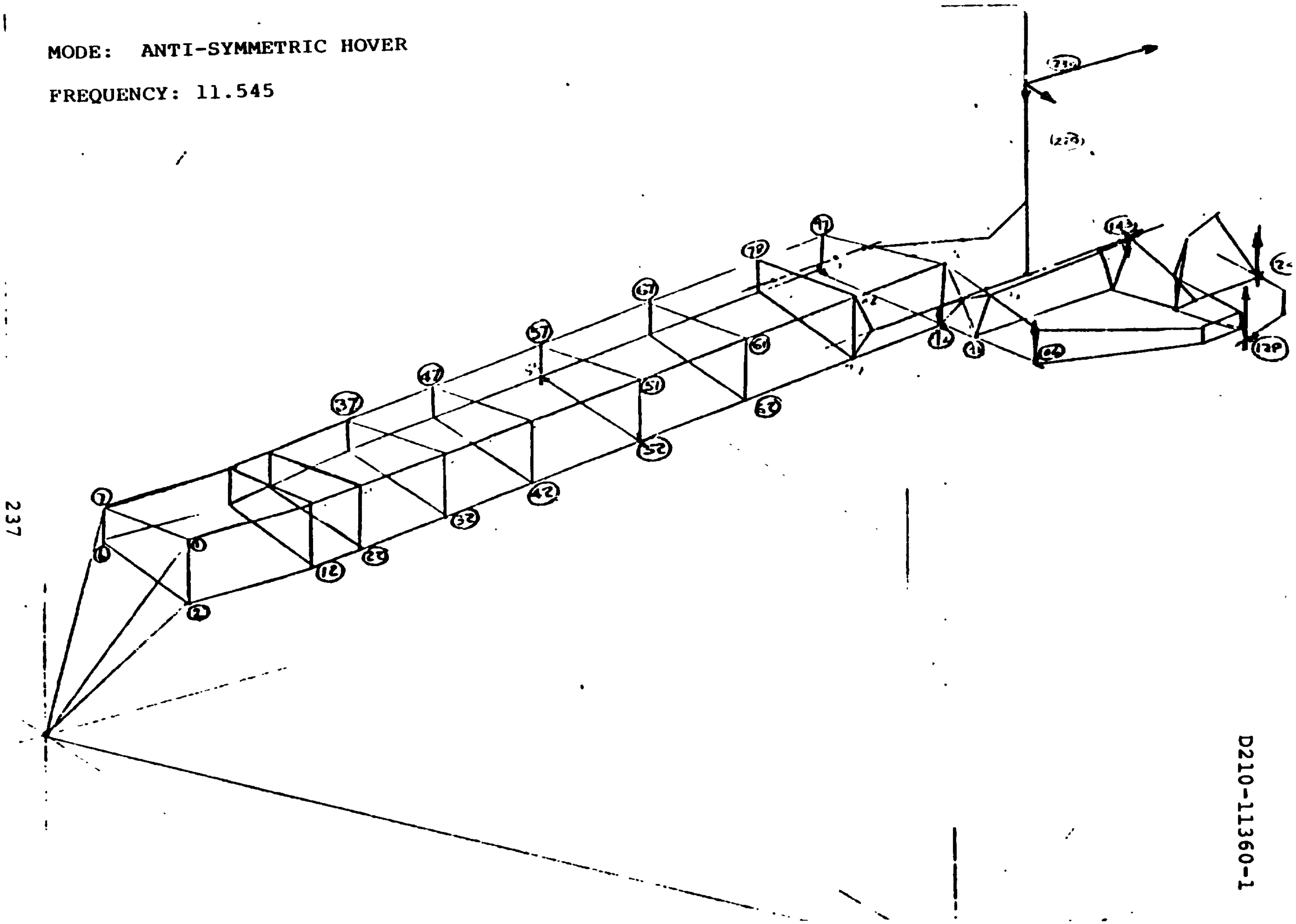


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC HOVER

FREQUENCY: 5.836

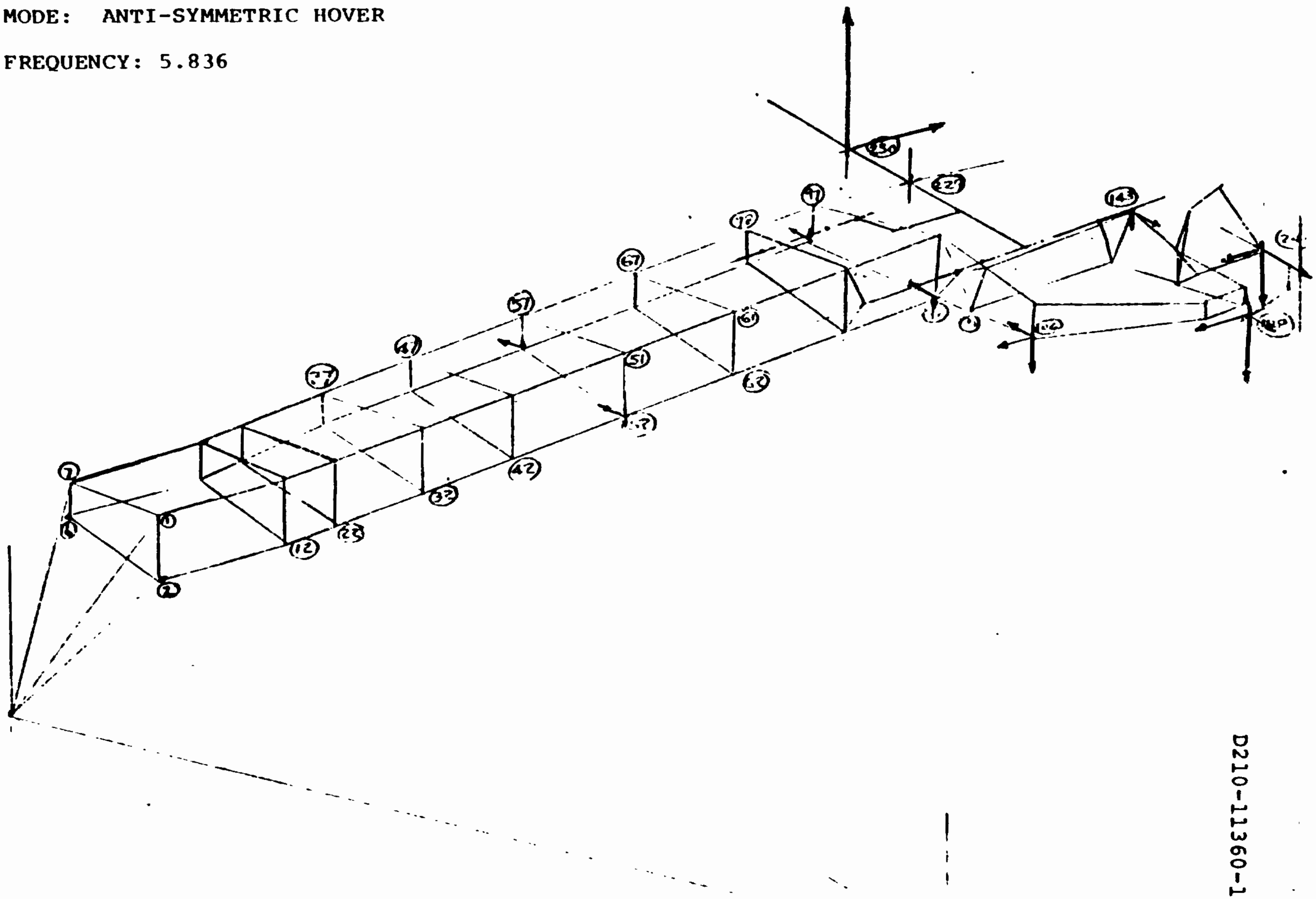


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC CRUISE

FREQUENCY: 6.651

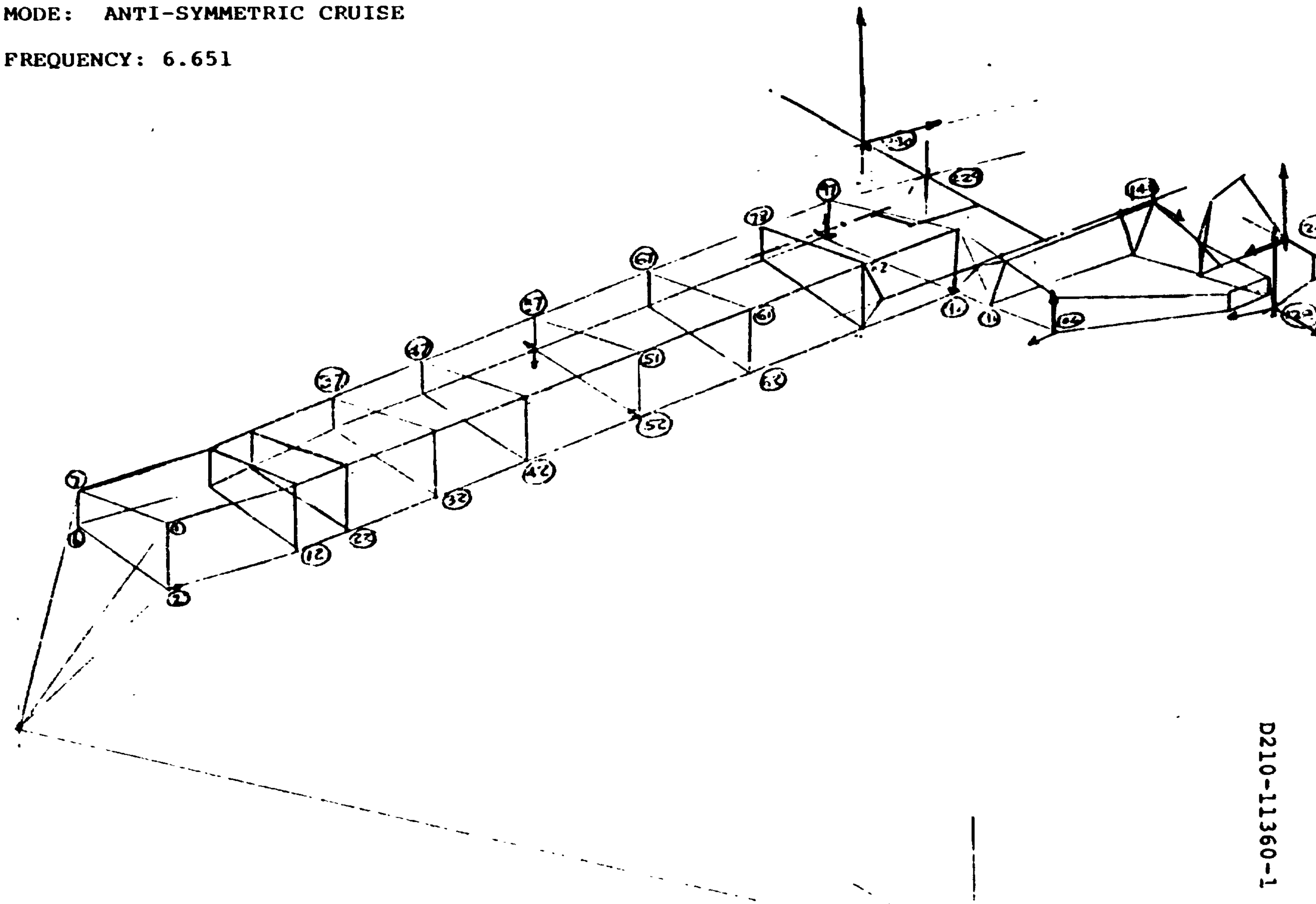


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC CRUISE

FREQUENCY: 9.191

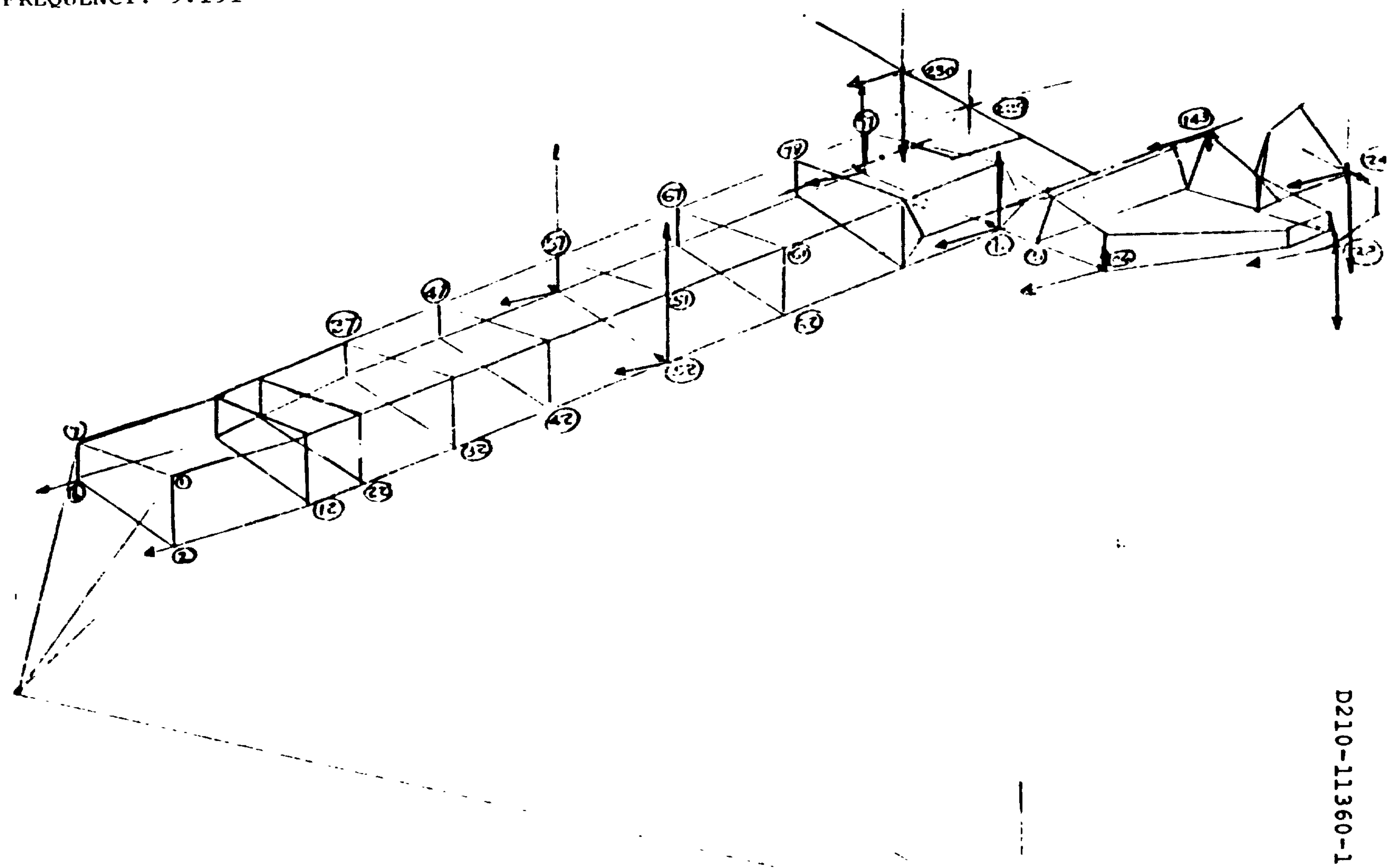


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

MODE: ANTI-SYMMETRIC CRUISE

FREQUENCY: 10.923

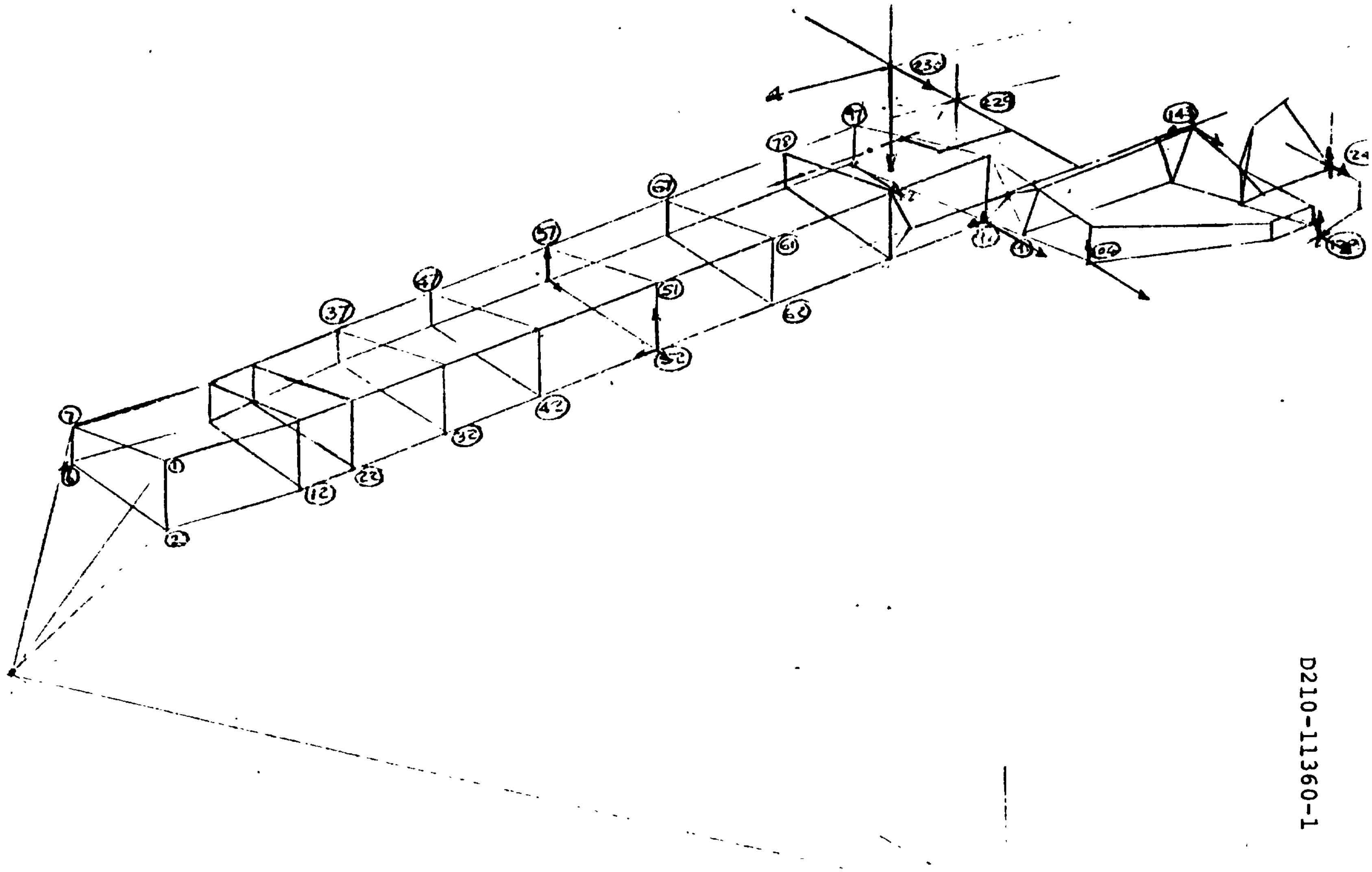


FIGURE 3.5.8. MODEL DISPLACEMENTS (CONTINUED)

3.5.3.2 Discussion

The data in Table 3.5 shows that the Boeing Vertol rotor system installation does not significantly affect the basic XV-15 wing natural frequencies. The approximately 10% reduction in wing bending frequencies is essentially due to two sources. The effective mass of the Boeing Vertol rotor system including engines and support structure is about 14% heavier than the XV-15 pylon mass per Reference 4, Table 5-3. Also, the rotor system mass center of gravity is located further outboard. The presence of an undesirable pylon pitch mode and reduction in the pylon yaw mode natural frequencies indicates presence of "softness" in the rotor nacelle trunnion and aft support structure areas. As previously stated, these areas have been progressively beefed up. Several iterations using selective beef-up of 'soft spots' in the local structure will be required before the design of the Boeing Vertol rotor system structure is finalized.

3.5.4 Conclusion

On the basis of the studies conducted during the course of this effort, it is concluded that

- The structural integrity of the basic XV-15 wing is not adversely affected by the modifications required to install the Boeing Vertol rotor system on the aircraft.
- A beef-up is required for the nacelle tilt actuation system.

- Selective stiffening of the Boeing Vertol rotor nacelle and aft support structure and a modification to the trunnion support concept are required to eliminate an undesirable local pylon pitch mode.

3.6 FLYING QUALITIES AND FLIGHT BOUNDARIES

This section presents an assessment of the flight-boundaries and flying qualities of the fixed-engine HTR XV-15. In Reference 1 the flying qualities of the existing XV-15 with its 25-foot gimballed rotors removed and replaced by Boeing hingeless 26-foot diameter rotors were examined in detail using a piloted simulation math model. The fixed-engine, hingeless rotor design for the XV-15 presented in this report differs from that evaluated in Reference 1 mainly (from a flying qualities standpoint) in that the weight and inertias are increased.

3.6.1 Control Positions and Aircraft Attitudes in Transition

The aircraft trimmed attitudes and control positions required in transition were computed at the design gross weight of 6154 kg (13,568 lb) at the aft c.g. position in hover - Station Line 7.65 m (301.2 inches). The results were obtained with the cyclic-on-the-stick blade load minimization system present. This system, developed as part of the work reported in Reference 1, applies cyclic control to the rotors as a function of longitudinal stick position and nacelle angle. These cyclic inputs are in addition to the n[illegible]l pilot control inputs. The load minimization cyclic schedules are presented in Figures 3.6.1 and 3.6.2. (The cyclic inputs from longitudinal stick continue in cruise).

HTR XV-15
CONTROL AXIS CYCLIC PITCH INPUT AS A FUNCTION OF LONGITUDINAL STICK AT $i_N = 0^\circ$

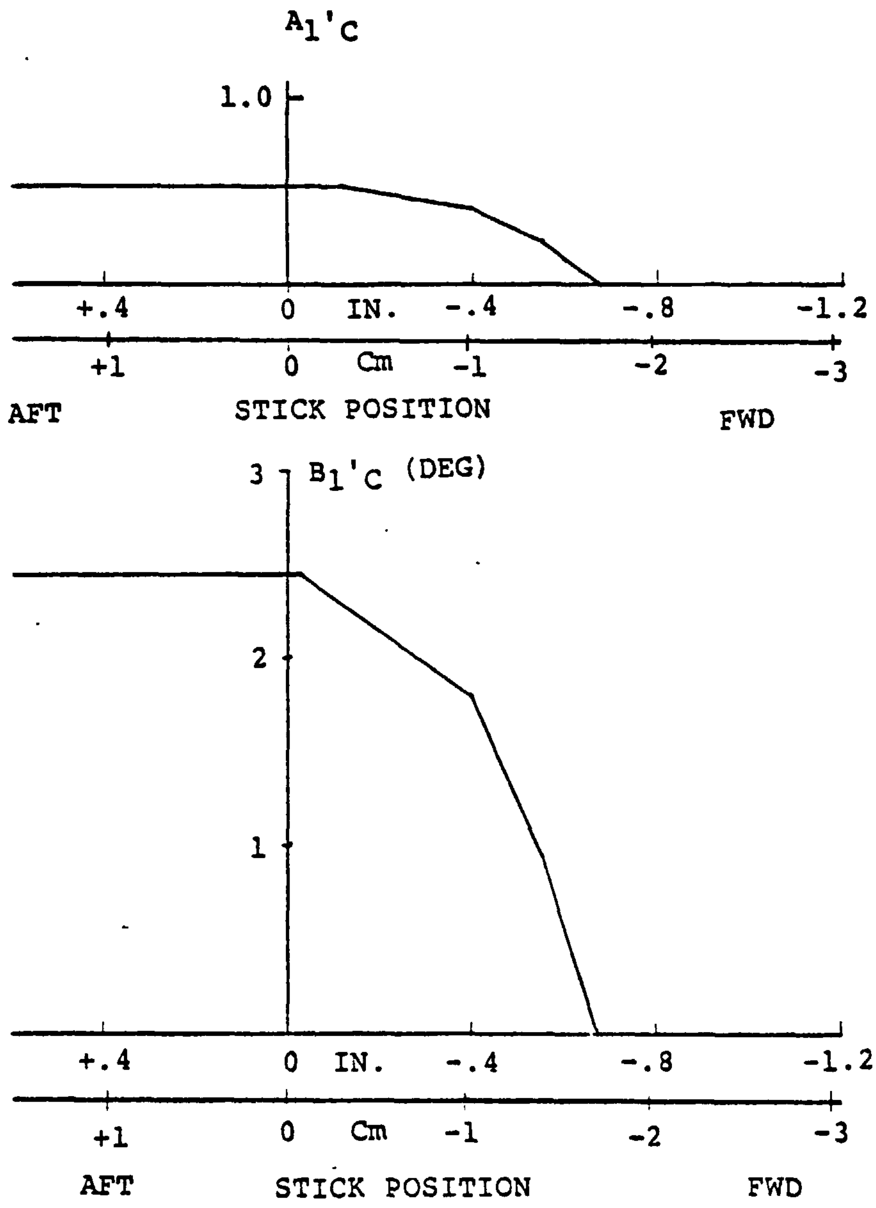
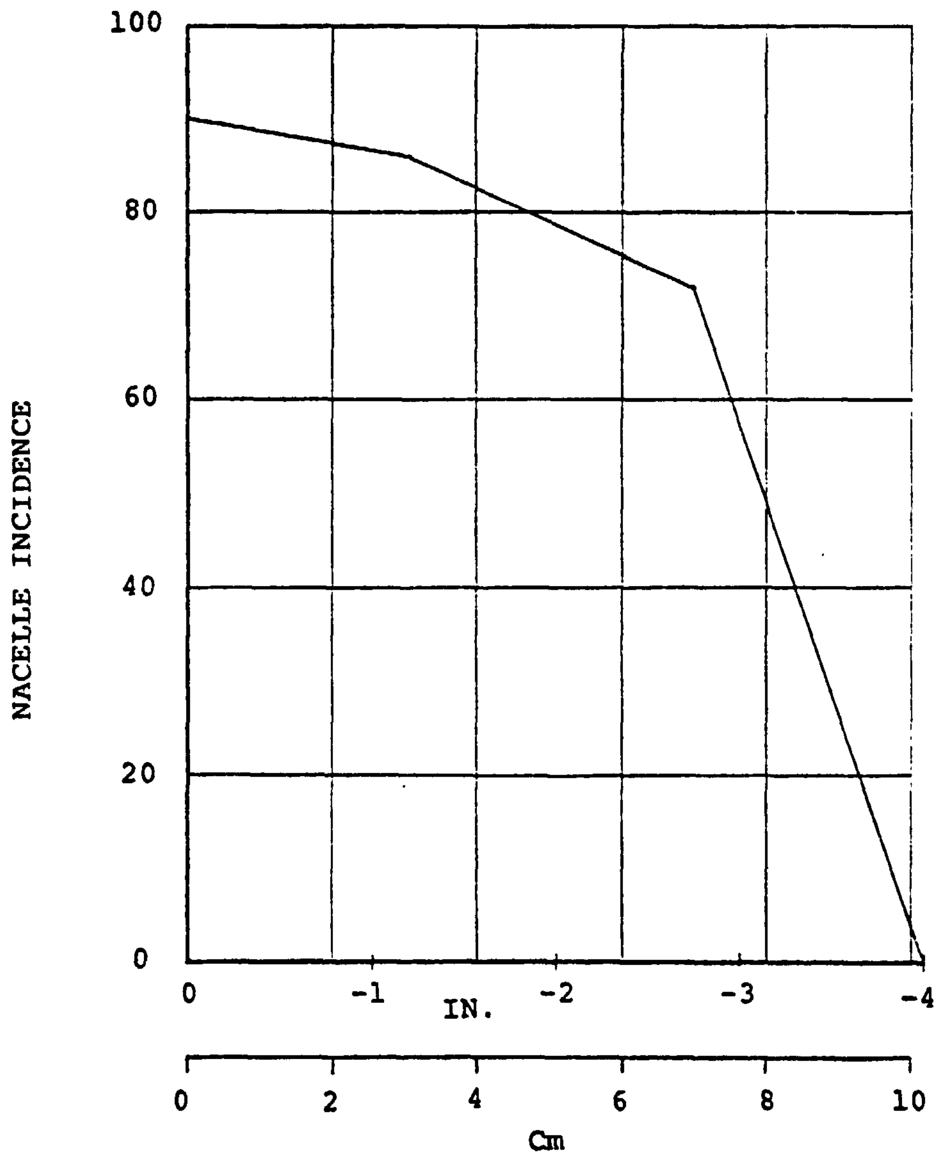


FIGURE 3.6.1. CYCLIC PITCH CONTROL ON THE STICK AT $i_N = 0^\circ$

HTR XV-15 - LONGITUDINAL STICK BIAS



LONGITUDINAL STICK BIAS FOR CYCLIC CONTROL

FIGURE 3.6.2.CONTROL SYSTEM LONGITUDINAL STICK BIAS

3.6.1.1 Steady Level Transition

The trimmed aircraft fuselage attitudes are presented in Figure 3.6.3 as a function of airspeed at different nacelle angles. The data was obtained with the flaps set at 40 degrees. The variation of fuselage angle is smooth and continuous at each nacelle setting. These pitch attitudes also correspond to the wing angles of attack for those portions of the wing not influenced by the rotor slipstream. The wing angles of attack for the slipstream-immersed portions, as computed by momentum theory, are shown in Figure 3.6.4.

Figure 3.6.5 presents the variation of longitudinal stick position with speed in transition for different nacelle settings. The gradients with airspeed are stable (forward stick with increasing airspeed) at all nacelle settings. Below 120 knots, for nacelle angles less than 60°, a steepening of the stick gradient takes place as stall is approached. Full stick travel is ± 4.8 inches. At high nacelle angles (75° to 90°) and high airspeed (120 knots) the full-forward stick travel is approached. However, these conditions require very large nose-down attitudes (see Figure 3.6.4) and will not be met during normal flight.

3.6.1.2 Blade Loads in Transition

Calculated alternating bending moments at 12.5% of blade radius are presented in Figure 3.6.6. The bending moments were estimated using an empirical blade loads equation (Reference 1) that predicts balde loads to within 20% of these measured in

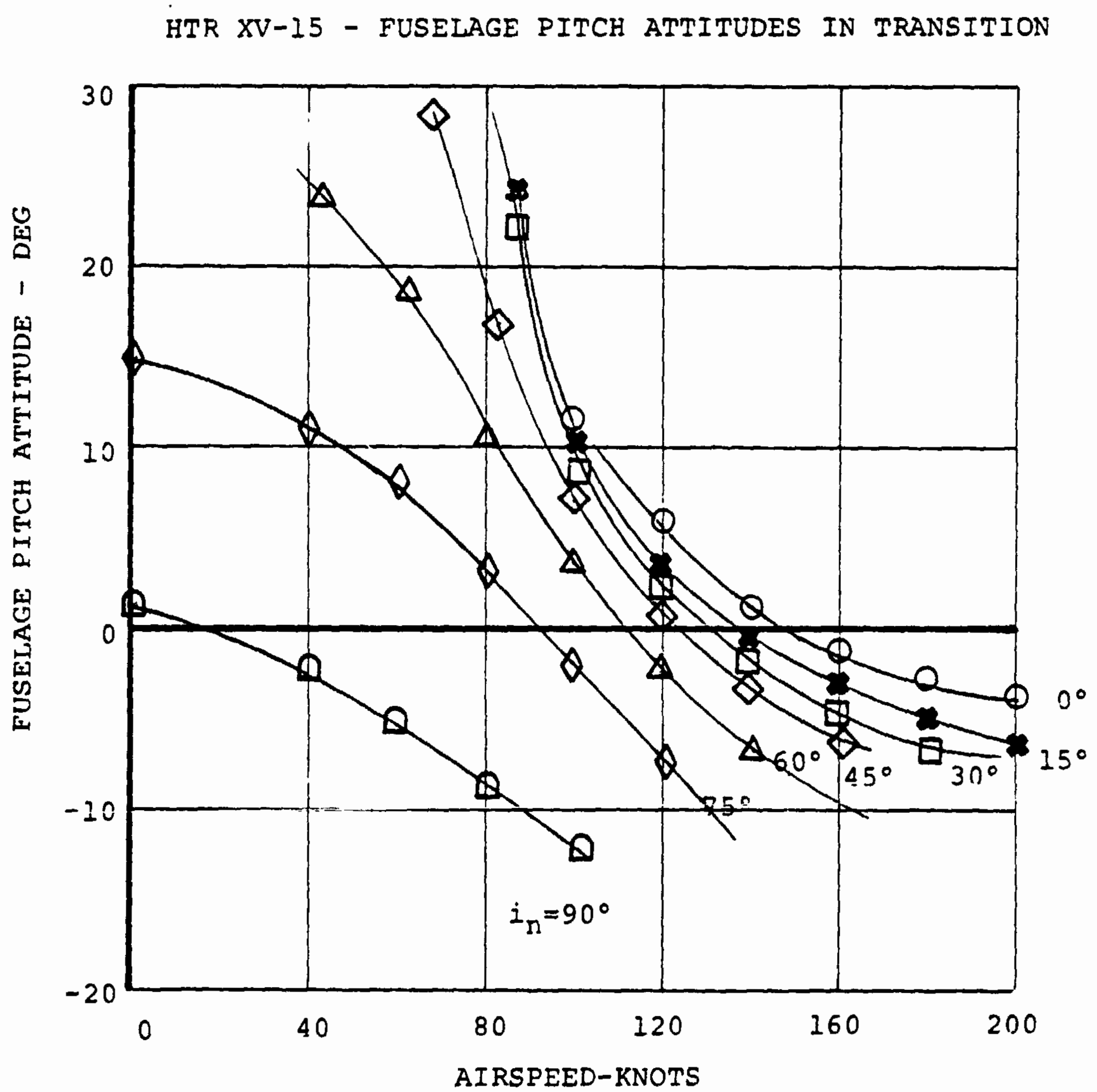


FIGURE 3.6.3. FUSELAGE PITCH ATTITUDE IN TRANSITION
AFT CG, SL STD, GW = 6154 KG (13,568 LB)
FLAPS = 40°

HTR XV-15 - WING ANGLES OF ATTACK THROUGH TRANSITION

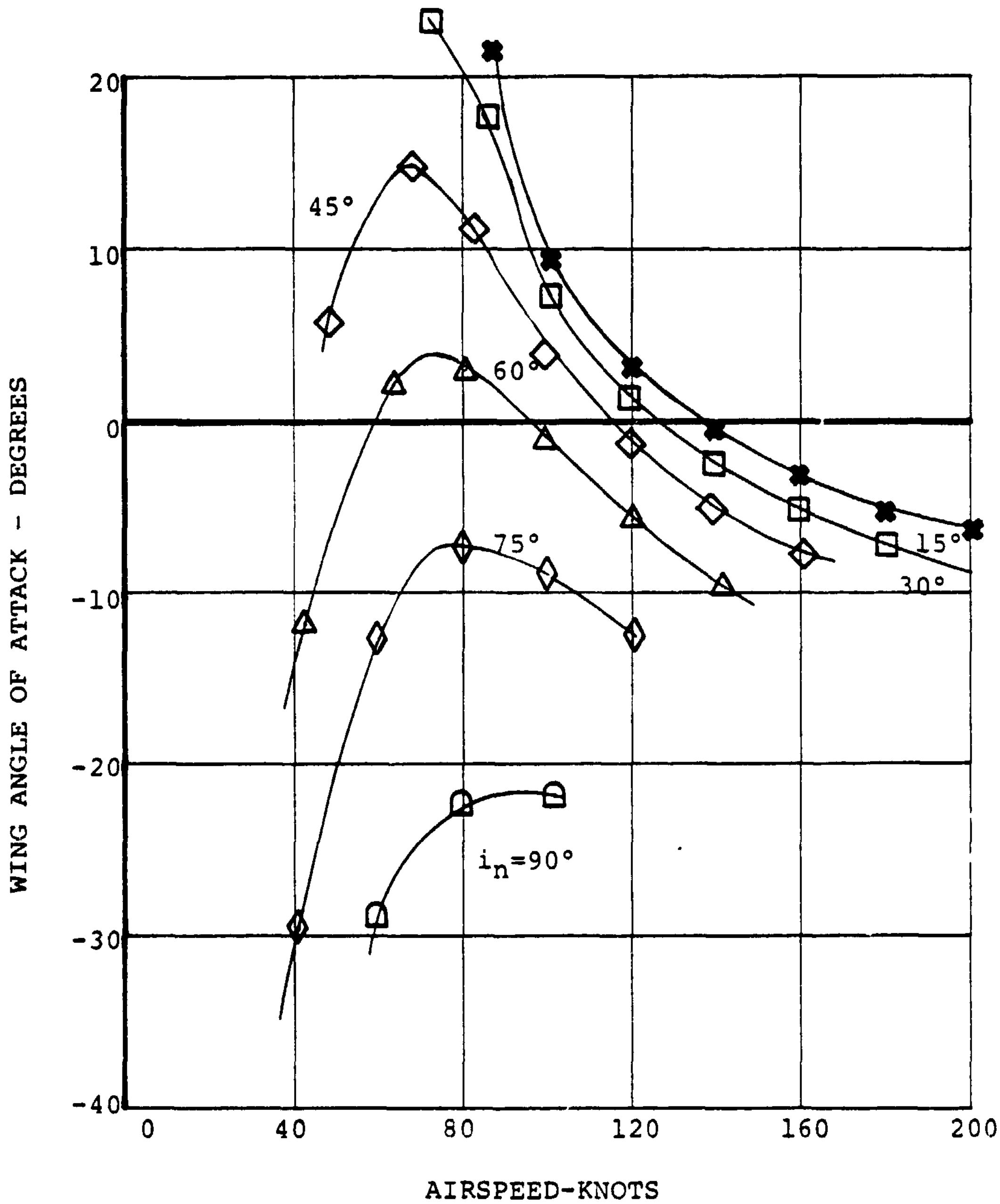


FIGURE 3.6.4.WING ANGLE OF ATTACK IN SLIPSTREAM
AFT CG, GW = 6154 KG (13,568 LB) SL STD

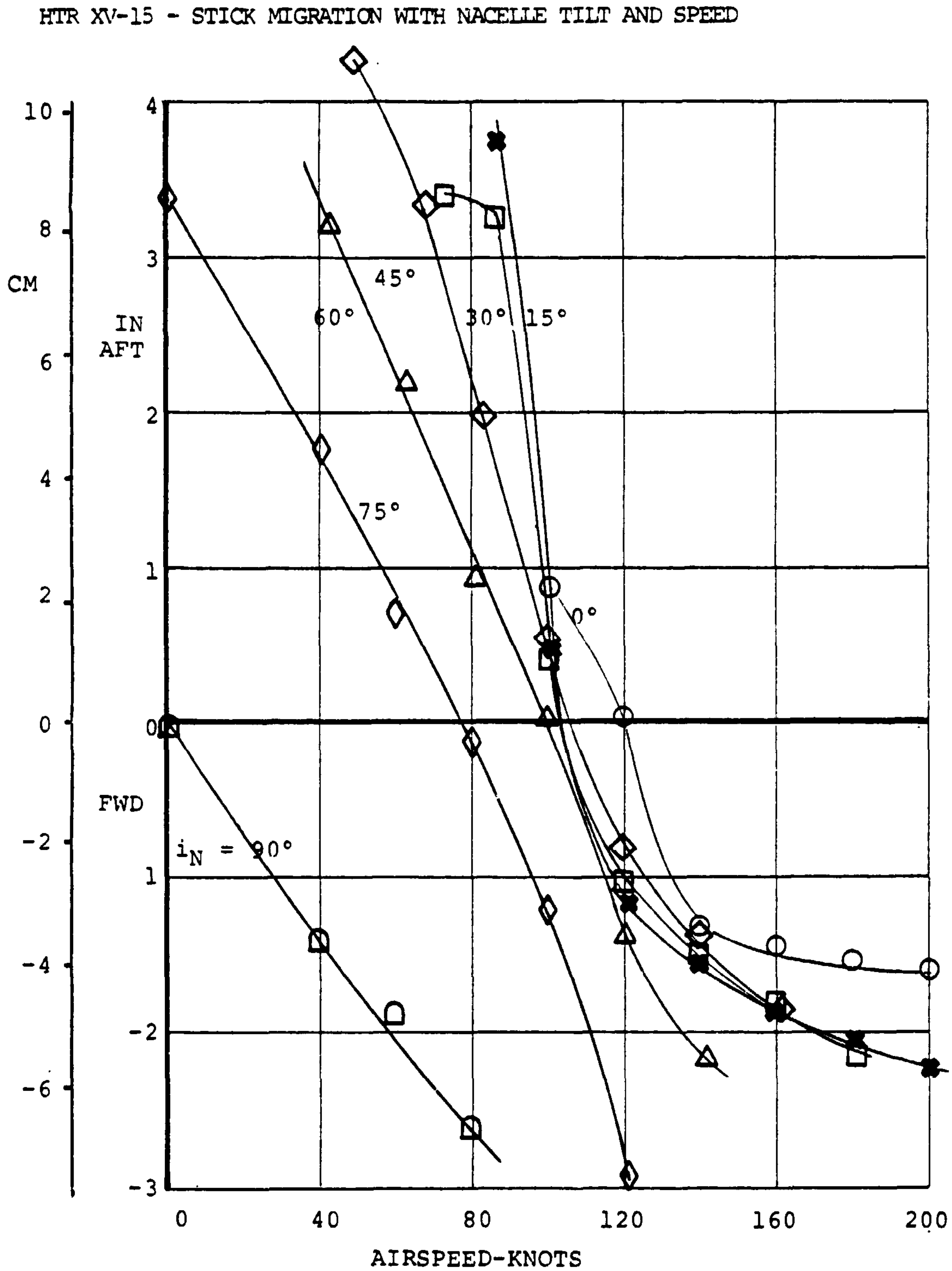


FIGURE 3.6.5.LONGITUDINAL STICK POSITION - AFT CG SL STD,
GW = 6154 KG (13,568 LB) FLAPS = 40°

HTR XV-15 - BLADE LOADS IN TRANSITION

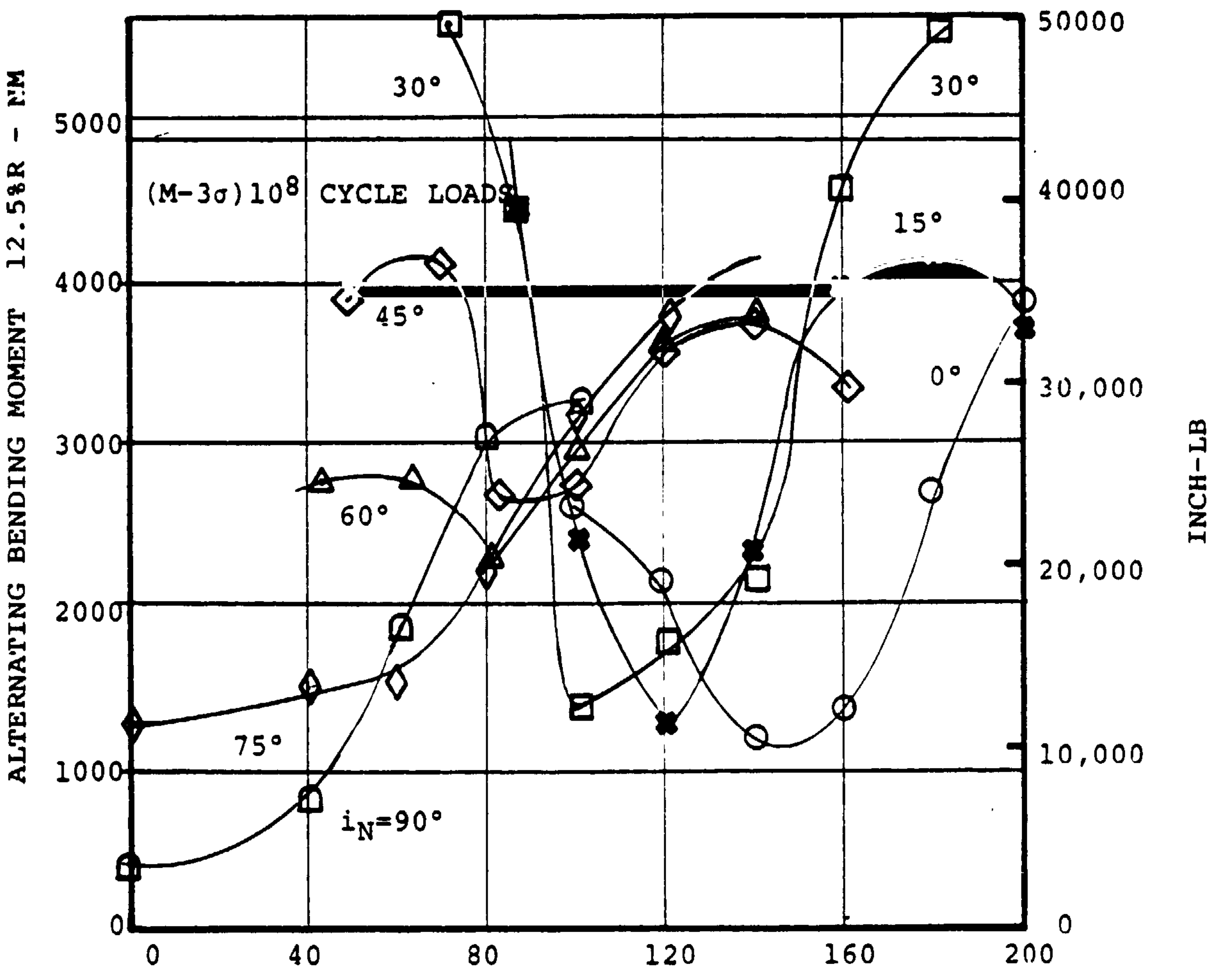


FIGURE 3.6.6. ESTIMATED BLADE BENDING LOADS IN TRANSITION
AFT CG SL STD, GW = 6154 KG (13,568 LB)
FLAPS = 40°

tests of the full-scale rotor in the Ames 40-foot by 80-foot tunnel. With the present blade design, a blade load level of 4888 NM (43,300 in.-lbs) corresponds to the calculated infinite fatigue life allowable. This level is indicated in Figure 3.6.6. The blade loads are sufficiently low throughout transition that an acceptable flight corridor exists which is free of blade fatigue limitations.

3.6.1.3 Transition Corridor

The estimated transition corridor for the HTR XV-15 at design gross weight is presented in Figure 3.6.7. The corridor is bounded by wing stall at low airspeeds. The stall boundary shown was obtained from Figure 3.6.4, and assumes that the slipstream does not delay the occurrence of wing stall as would be indicated from momentum calculations. Tests have s own that at high nacelle angles above 40 knots, the rotor slipstream leaves the wing and blows back along the free stream. Thus, above 40 knots the wing angle of attack is more reliably obtained from the fuselage attitude.

At the high speed end of the transition corridor, the allowable transmission torque limits flight above 60 degrees nacelle angle. This boundary also coincides with maximum available down-elevator deflection. Between 20 and 60 degrees nacelle setting the blade loads boundary is encountered before the torque limit is reached. Below 20° the flap dynamic pressure limits flight to 170 knots. In helicopter flight an airspeed band of some 130 knots is usable, while in the aeroplane

HTR XV-15 - LIMITS ON TRANSITION CORRIDOR

GW = 6154 kg (13568 lb)
AFT CG
S.L. STD DAY
δ_f = 40°

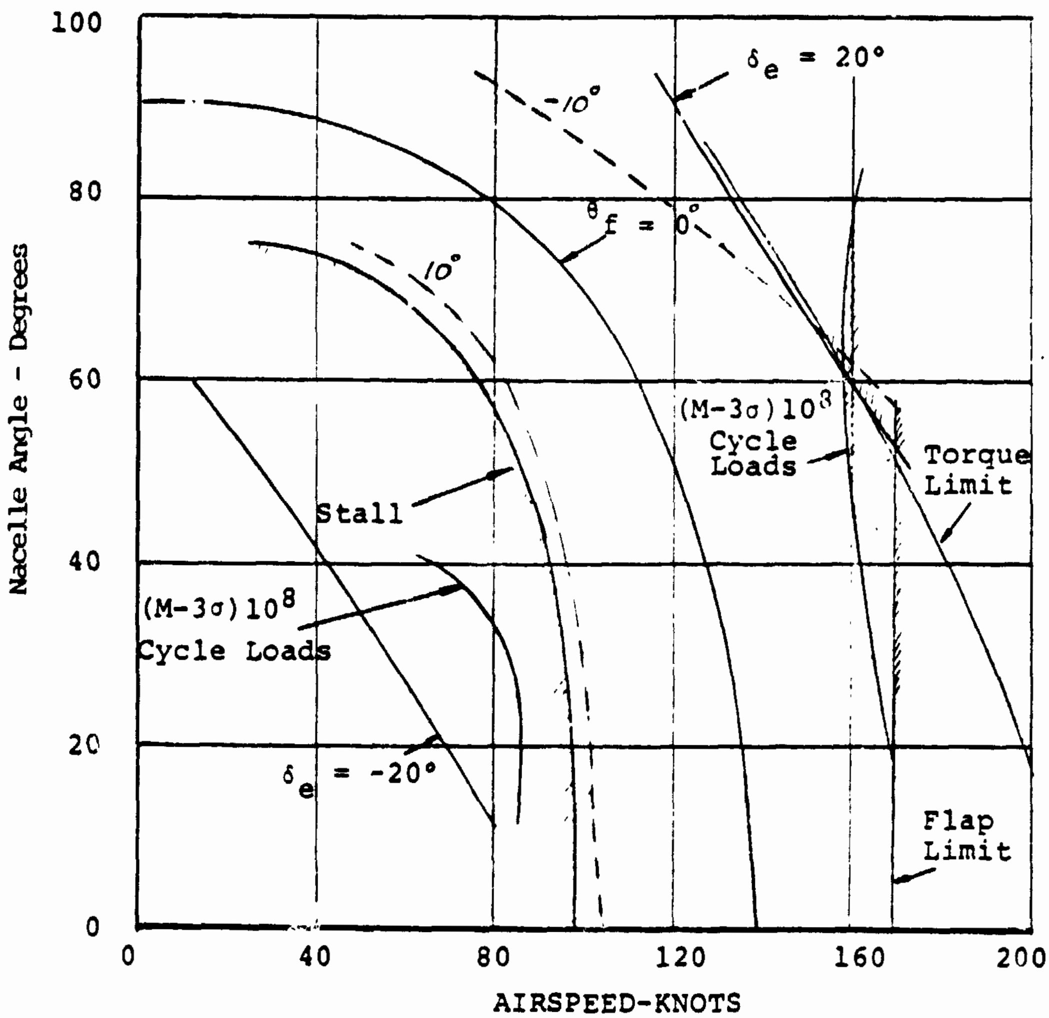


FIGURE 3.6.7. TRANSITION CORRIDOR - AFT CG

conversion mode (i_N < 75°) a 70 knot range is available.

3.6.1.4 Coordinated Turns in Transition

The control positions, blade loads and power required to maintain coordinated turns are presented in Figures 3.6.8 through 3.6.13 for two points in the transition corridor corresponding to level fuselage attitude at zero bank angle.

At 80 knots and 75 degrees nacelle angle, the turn performance (ϕ = 65°) is limited only by the allowable transmission torque (1490 SHP at 551 RPM). At 120 knots with the nacelles placed at 45°, a bank angle of 70° is attainable before encountering the torque limit and the blade fatigue limit. Thus, a turn capability in excess of 2g is available for the HTR XV-15 during transition.

3.6.2 Maneuver Performance in Cruise Flight

The maneuver performance in cruise flight with flaps up is presented in Figure 3.6.14 at the design gross weight 6154 kg (13,568 lb) at sea level. The maneuver envelope is limited by wing stall and transmission torque. At 185 knots the blade fatigue load boundary touches the stall and torque boundaries. Elsewhere blade loads do not set a limit to turn performance. Maximum turn performance occurs at 186 knots where 2.5 g's can be achieved.

3.6.3 Control Power

The control power of the HTR XV-15 was evaluated over the airspeed range from hover through cruise. The control powers

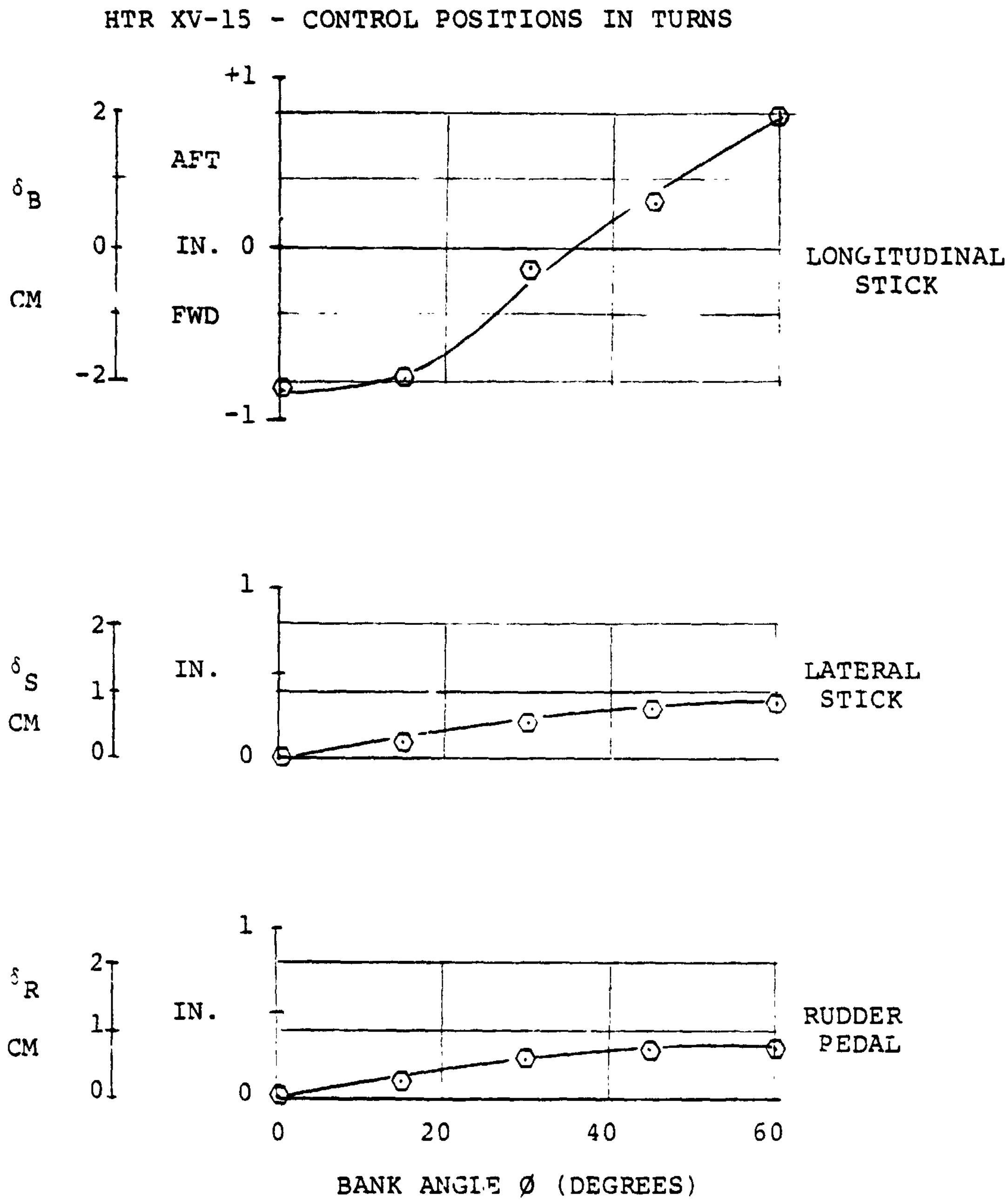


FIGURE 3.6.8. CONTROL POSITIONS IN COORDINATED TURNS IN TRANSITION, AFT CG, V = 120 KNOTS, i_N = 45°, GW = 6,154 KG (13,568 LBS), SL STD DAY, δ_F = 40°

HTR HV-15 - BLADE LOADS IN TURNS

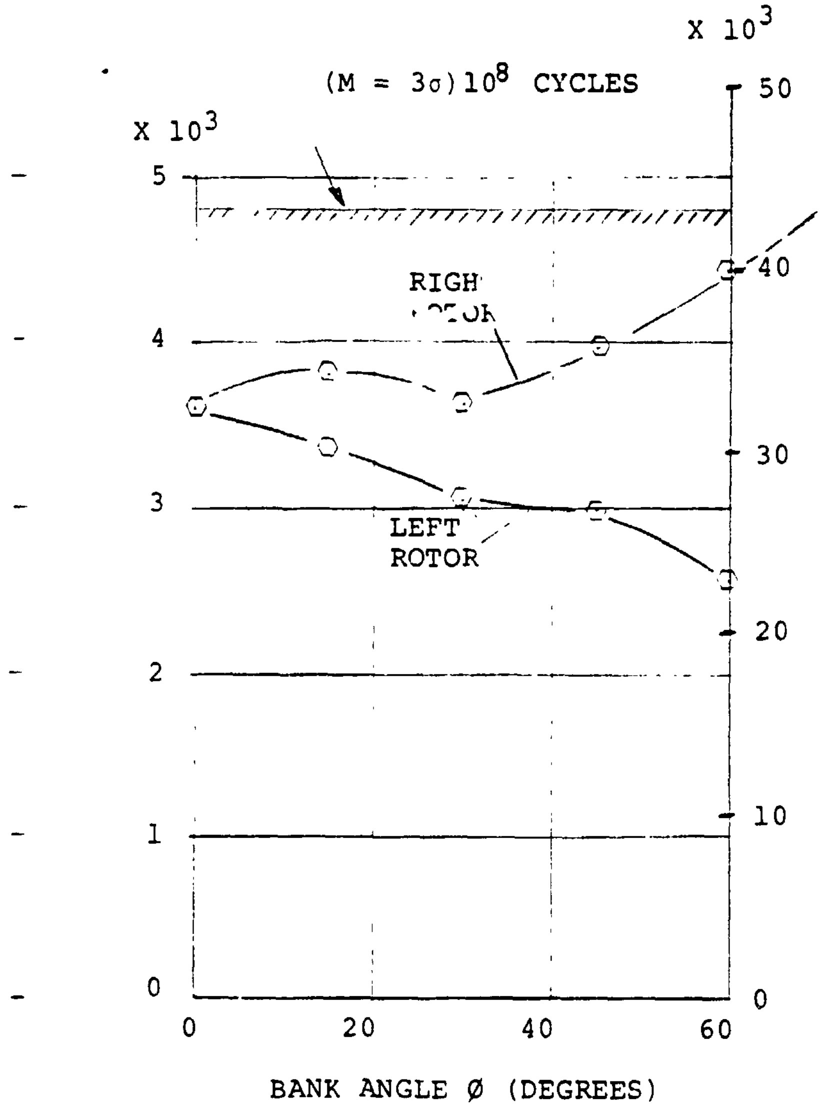


FIGURE 3.6.9. ESTIMATED BLADE BENDING LOADS, 12.5% IN COORDINATED TURNS IN TRANSITION, AFT CG, $i_N = 45°$, V = 120 KNOTS, $\delta_F = 40°$, GW = 6,154 Kg (13,568 LBS), SL STD DAY

HTR XV-15 - POWER REQUIRED IN TURNS

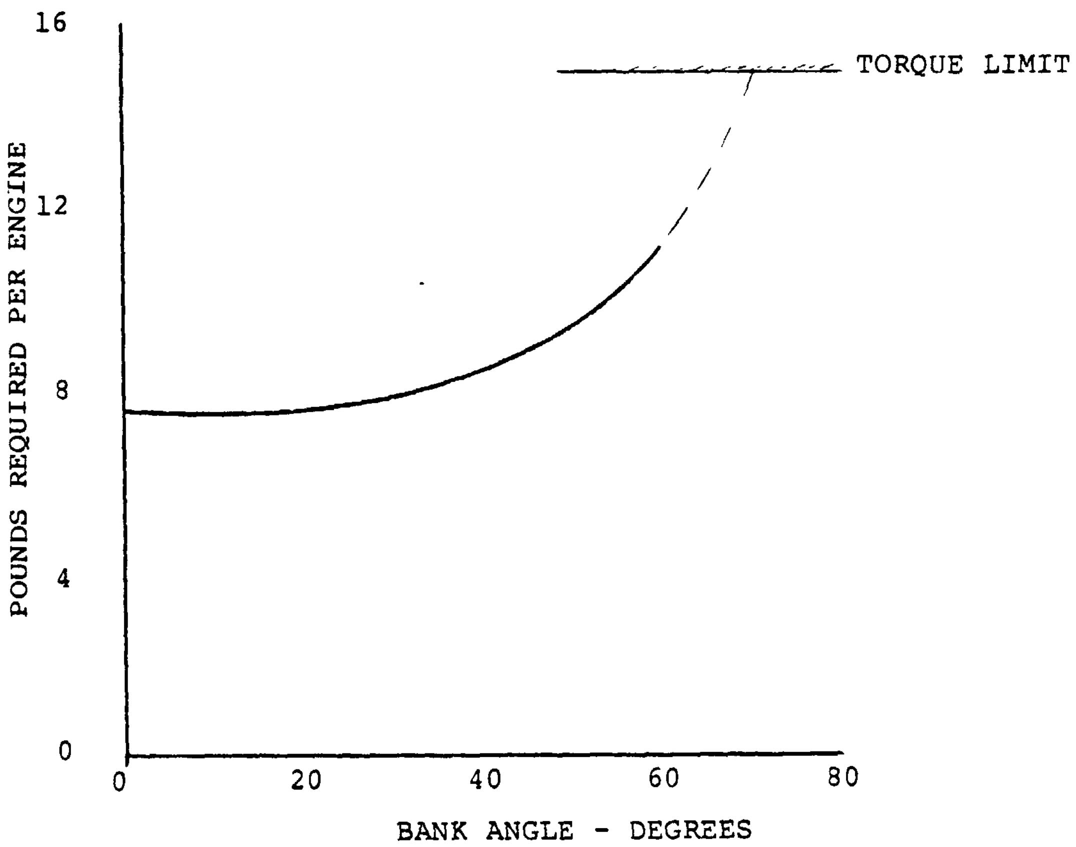


FIGURE 3.6.10. POWER REQUIRED IN COORDINATED TURNS IN TRANSITION, AFT CG, $i_N = 45°$, V = 120 KNOTS, SEA LEVEL, STANDARD DAY, 6,154 KG (13,568 LBS)

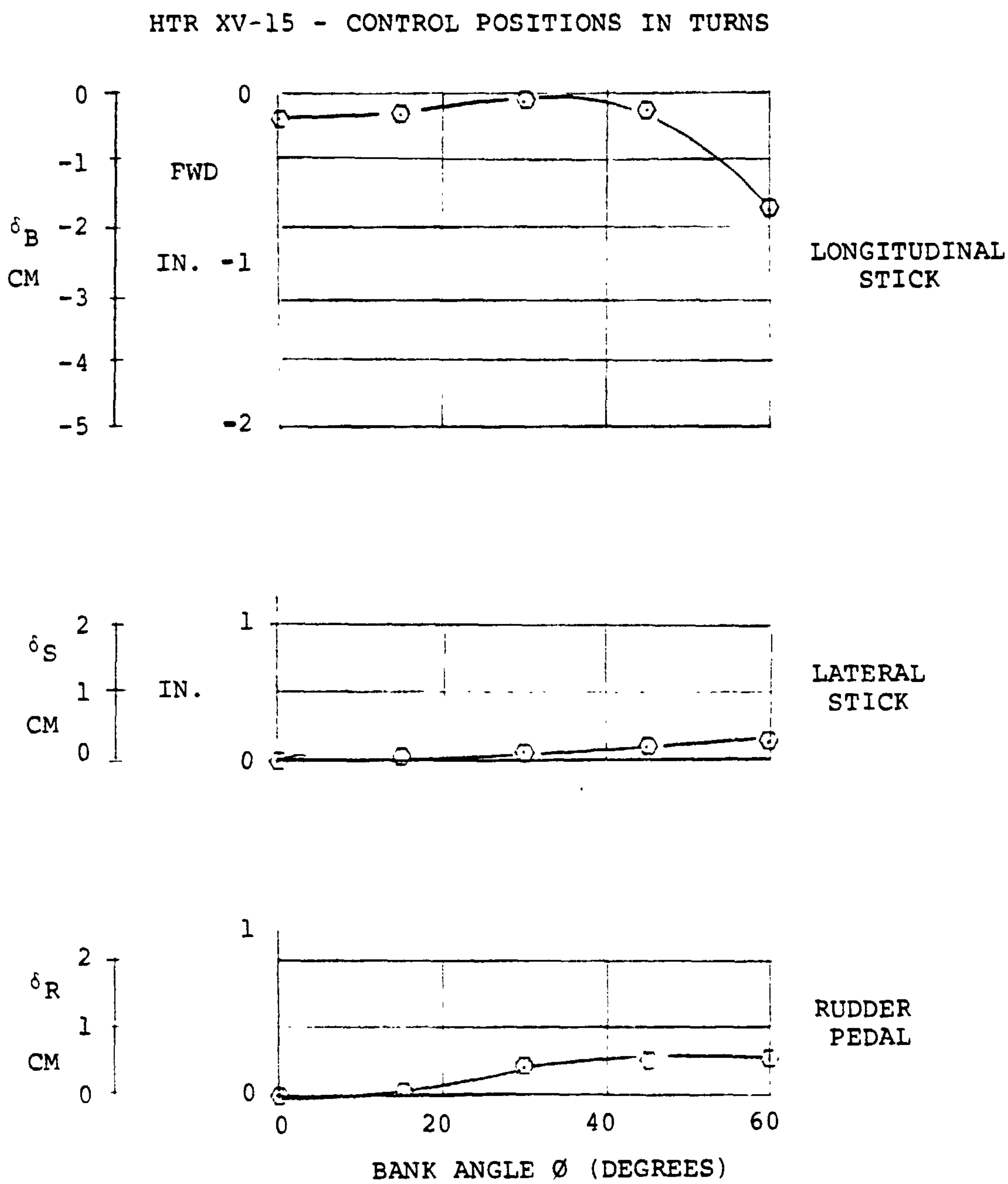


FIGURE 3.6.11 CONTROL POSITIONS IN COORDINATED TURNS IN TRANSITION, AFT CG, V = 80 KNOTS, i_N = 75°, GW = 6,154 KG (13,568 LBS), SL STD DAY, δ_F = 40°

HTR XV-15 - BLADE LOADS IN TURNS

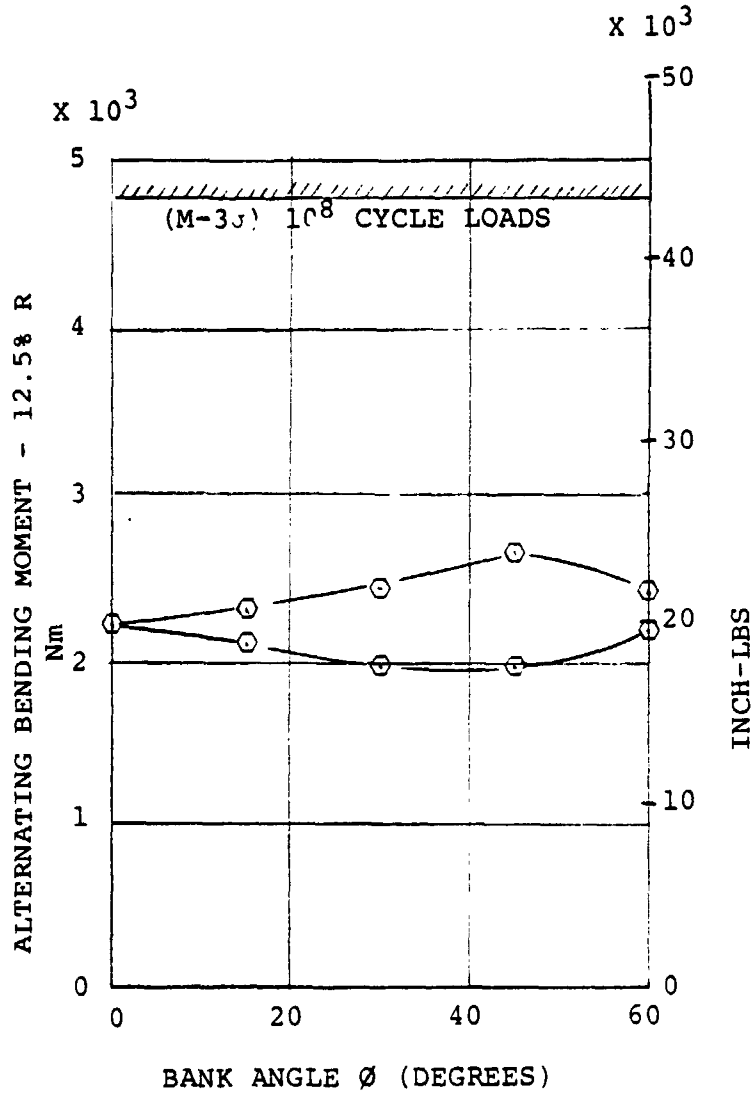


FIGURE 3.6.12 ESTIMATED BLADE BENDING LOADS, 12.5% IN COORDINATED TURNS IN TRANSITION, AFT CG, i_N = 75°, V = 80 KNOTS, δ_F = 40°, GW = 6,154 KG (13,568 LBS), SL STD DAY

HTR XV-15 - POWER REQUIRED IN TURNS

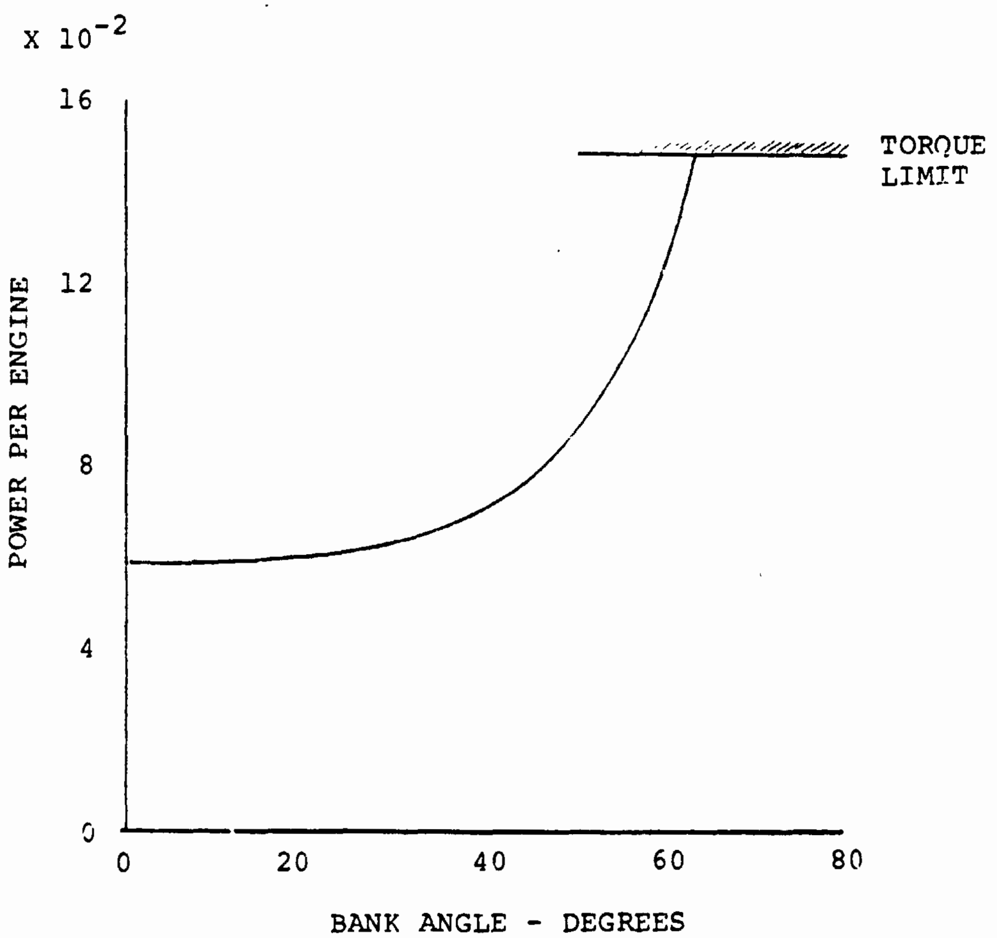


FIGURE 3.6.13. POWER REQUIRED IN COORDINATED TURNS IN TRANSITION, AFT CG, i_N = 75°, V = 80 KNOTS, SEA LEVEL, STANDARD DAY

HTR XV-15 - SUSTAINED TURN LIMITATIONS IN CRUISE

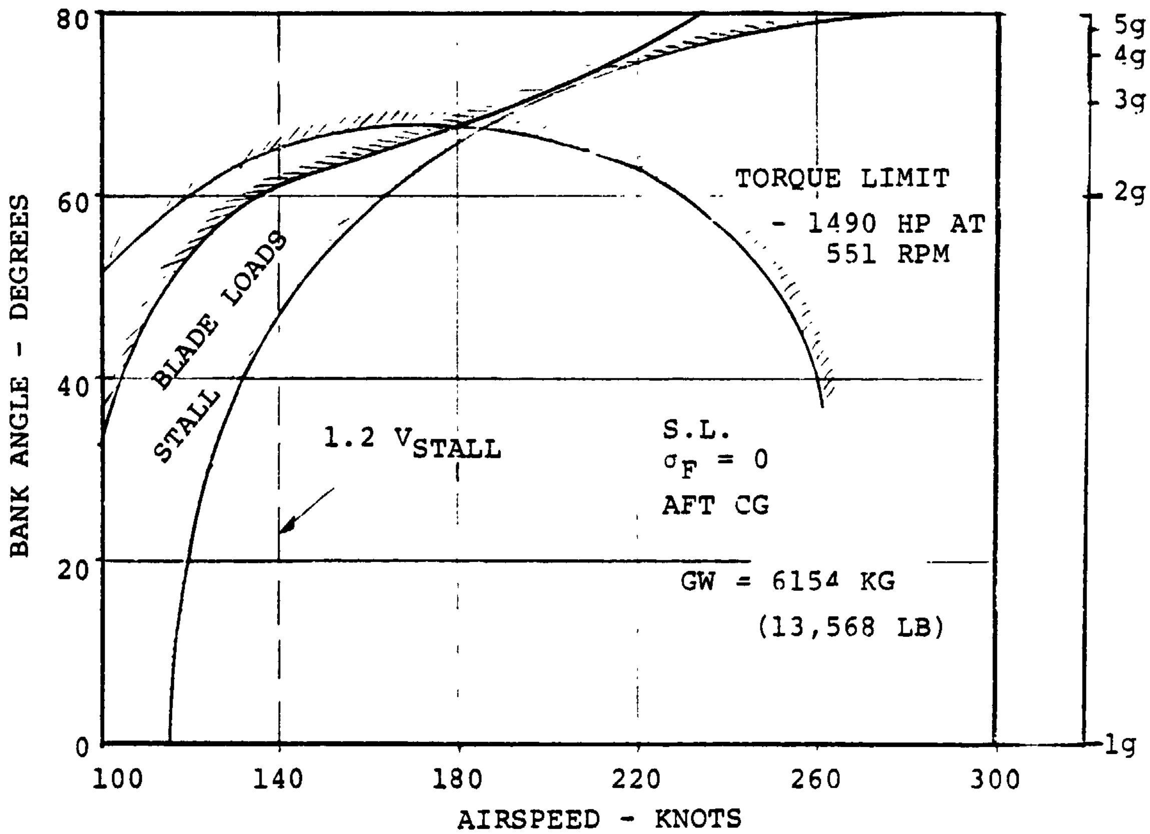


FIGURE 3.6.14.SUSTAINED-TURN PERFORMANCE IN CRUISE

were calculated for the aircraft at the design gross weight with the stability and control system inoperative. Pitch, roll and yaw control powers are presented in Figures 3.6.15, 3.6.16 and 3.6.17.

In hover, the control powers about all three axes are high and well above the minimums specified by MIL-F-93300. Pitch control power is in excess of the Level 1 requirement at all nacelle angles throughout conversion and into cruise. Roll control power in transition exceeds the requirement for Level 1 until a nacelle angle of about 45 degrees is reached. Thereafter, for the usable transition corridor, Level 2 minimums are met. Yaw control power is about the Level 2 minima except at 60 degrees nacelle angle and 100 knots. However, both these points are close to the stall boundary for the aircraft.

HTR XV-15 - PITCH CONTROL POWER

SYMBOL	i_N°	δ_N°
⬡	90	40
◊	75	40
△	60	40
□	30	40
X	0	40
○	0	0

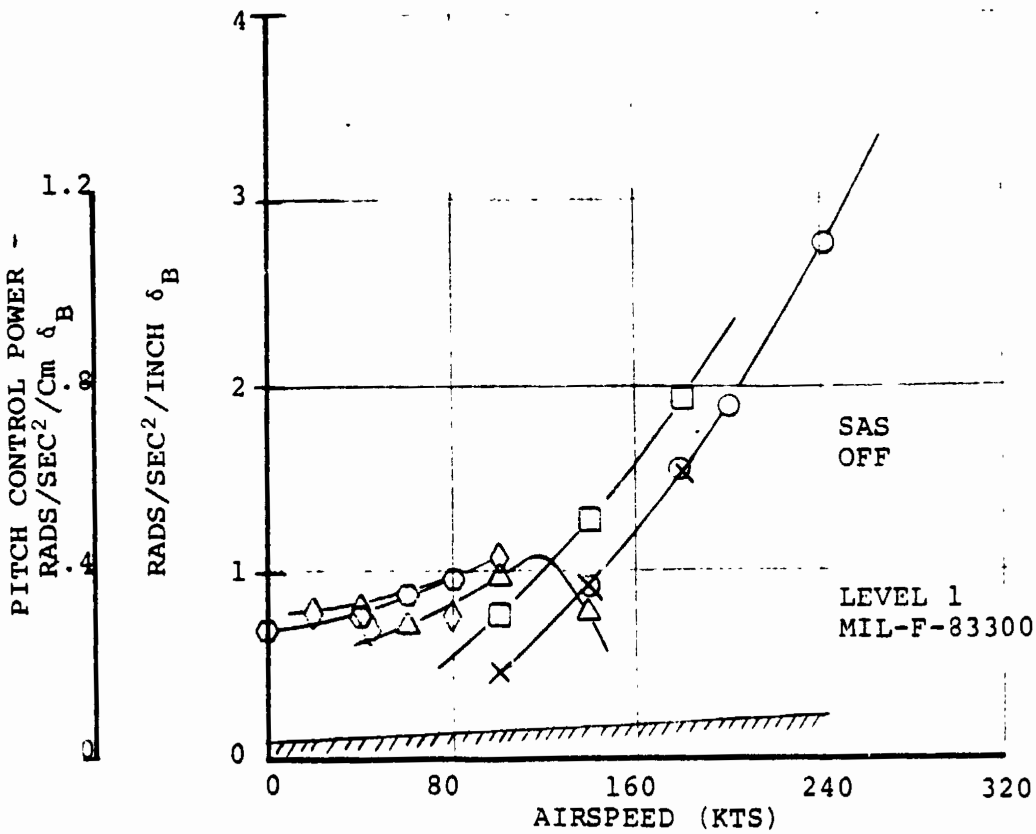


FIGURE 3.6.15. PITCH CONTROL POWER - AFT CG

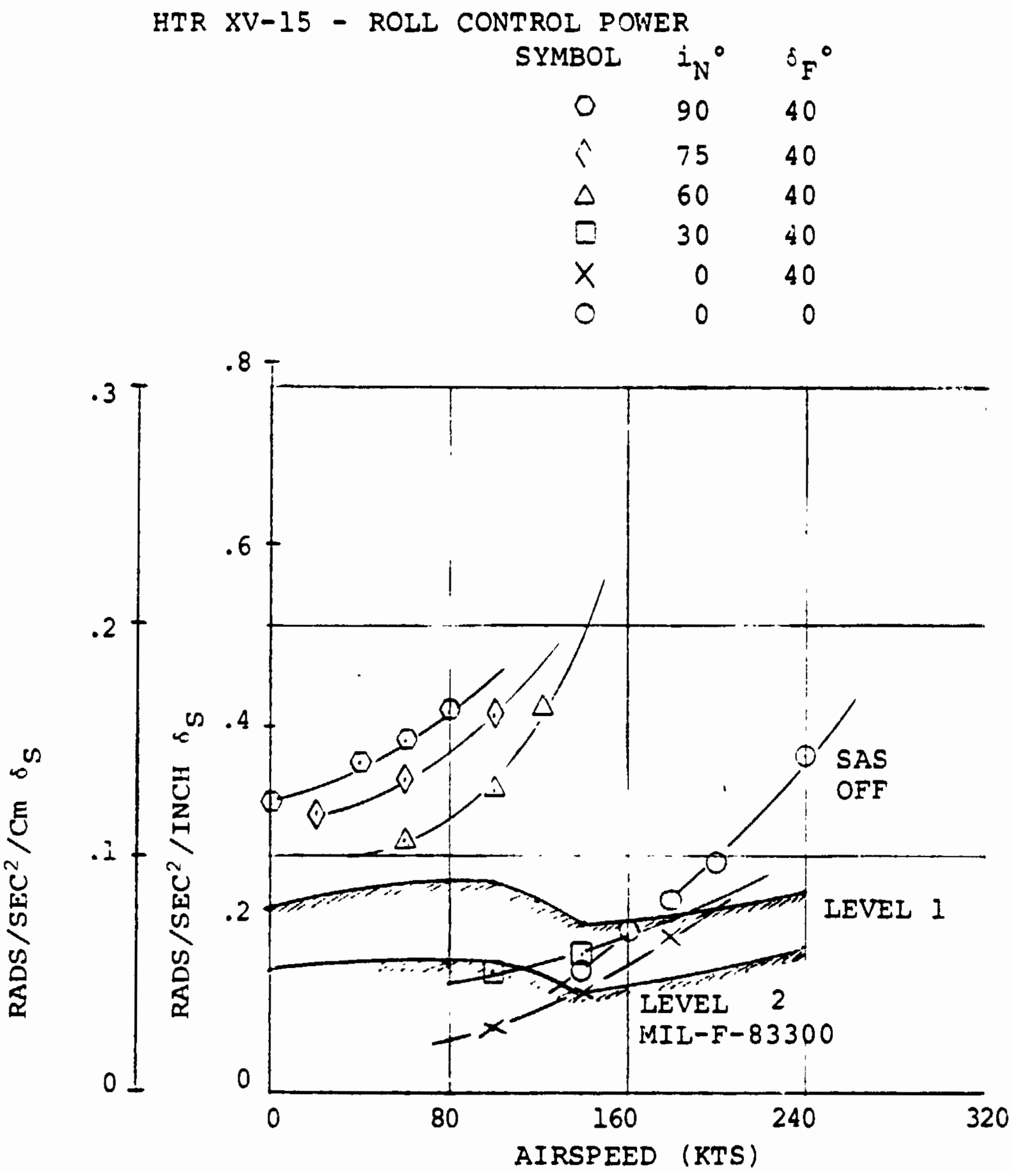


FIGURE 3.6.16. ROLL CONTROL POWER - AFT CG

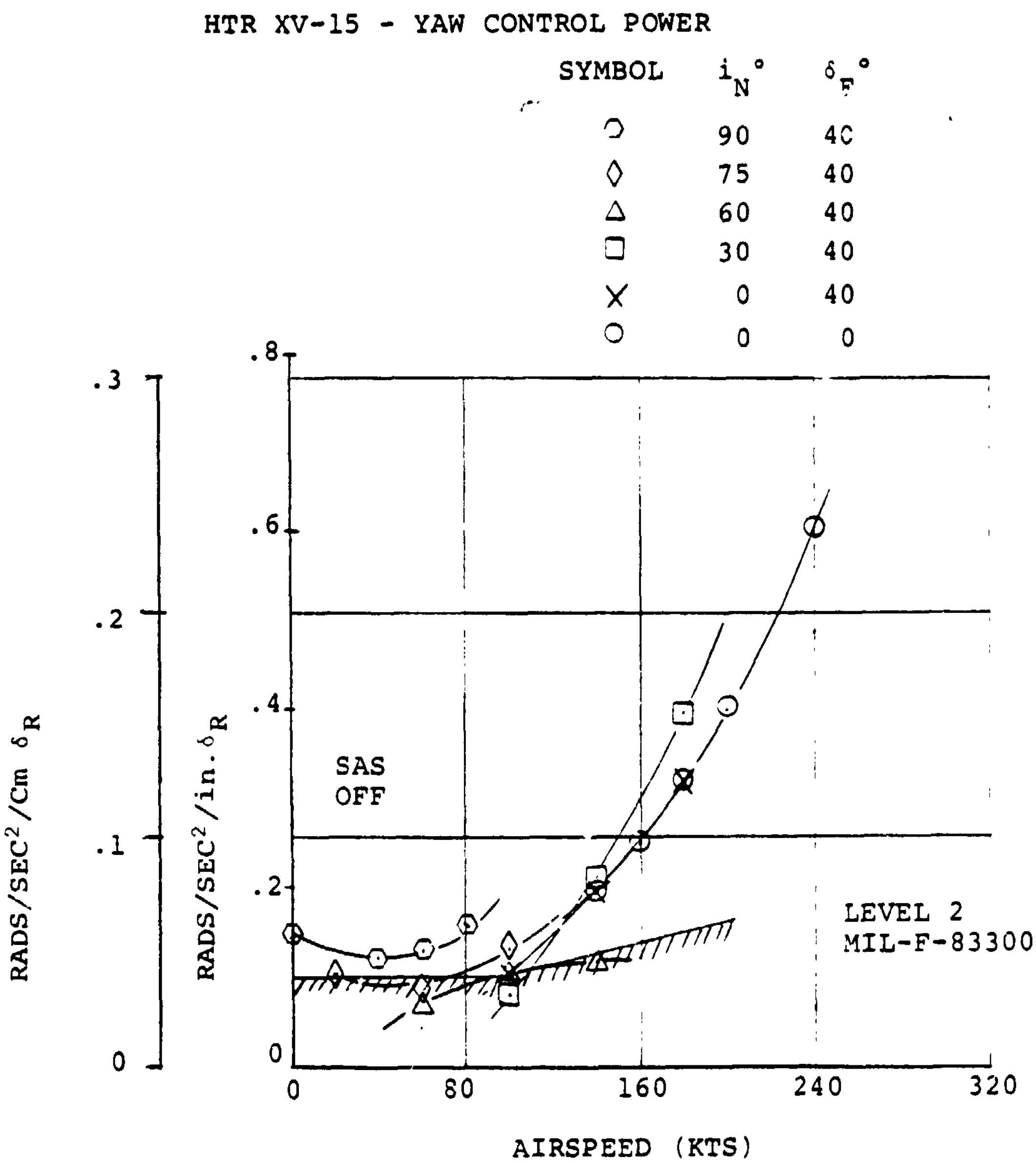


FIGURE 3.6.17. YAW CONTROL POWER - AFT CG

4.0 CONCLUSIONS

The design and evaluation studies reported in sections 2.0 and 3.0 of this volume allow the following general conclusions to be drawn.

1. The design of a new nacelle and rotor system to fly a hingeless soft in plane rot- and fly by wire controls on the XV-15 aircraft is feasible and a preliminary design definition has been provided.
2. The aircraft weight will increase by approximately 568 lbs. Most of this increase is attributable to the larger rotor, higher capacity transmission and the structure necessary to adapt a fixed engine configuration to a wing designed for a tilting engine configuration.
3. The fly by wire control system can be designed for this application and will provide a much greater degree of flexibility, higher reliability and a higher order of redundancy and safety than the mechanical system for about the same weight. Several additional problems are solved by the installation; for examp' .e potential mechanical interference of wing deflec .. on the control runs is not a problem with fly by wire.
4. The existing XV-15 tilt actuator can be u. . . either same modification or a limit on the aircraft control power.
5. The aircraft resulting from this modification will have higher cruise speeds, more hover lift capacity and better rates of climb than the existing XV-15.

6. The modified aircraft is free from aeroelastic restrictions on its normal flight envelope and has an essentially infinite fatigue life rotor system.
7. The control system is easily adapted for the addition of gust alleviation systems and other adaptive systems for future research.

5.0 REFERENCES

1. NASA CR-151950, Preliminary Simulation of an Advanced Hingeless Rotor XV-15 Tilt Rotor Aircraft, M. A. McVeigh, December 1976.
2. NASA TMX-62,407, NASA/ARMY XV-15 Tilt Rotor Research Aircraft Familiarization Document.
3. NASA CR-114664, V/STOL Tilt Rotor Aircraft Study Wind Tunnel Tests of a Full Scale Hingeless Prop/Rotor Designed for the Boeing Model 222 Tilt Rotor Aircraft, J. P. Magee, H. R. Alexander, October 1973.
4. Bell Helicopter Company, Document Number 301-199-001, -002, -003 and -004, V/STOL Tilt Rotor Research Aircraft, Revision A, November 1974.
5. Boeing Document D222-10050-1, -2, -3 and -4, Study of V/STOL Tilt Rotor Research Aircraft Program, January 1973.
6. NASA CR-151937, Wind Tunnel Test on a 1/4.622 Scale, Hingeless Rotor, Tilt Rotor Model, September 1976.

APPENDIX I - TRADE STUDY DATA

AI-1 - CONFIGURATION STUDIES

Preliminary design studies were made of the configuration alternatives. The problem was to determine the best arrangement of a new XV-15 wing-tip nacelle appropriate for use with a Boeing Vertol hingeless rotor and fly-by-wire controls.

Initial Tradeoffs

A review was made of the wing-tip nacelle designs of the current XV-15 aircraft and of the Boeing Vertol Model 222 competitor aircraft. The selection of drive train arrangement so strongly influenced the overall nacelle design that drive configuration had to be determined first. Two basic considerations were whether the engine would stay fixed or tilt, and whether a single or two separate inputs (from engine and from wing cross shafting) to the rotor transmission should be used. Four options, A thru D, were roughly defined as summarized in Table A1.1 and portrayed in Figures A1.1 through A1.7.

A fixed (non-tilting) engine was selected over a tilting arrangement for the following reasons:

- Technical or Operational
  - In general, there is no need for engine design/qualification for vertical operation (though T53 model for XV-15 is so qualified).

C-4

- Engine exhaust is not directed at the ground thus minimizing possibilities of surface fire, deterioration, personnal hazard, or exhaust reingestion.
- No swivel joints are required in the nacelle for engine services.

- More nacelle ground clearance for a given aircraft roll angle.

Options (C) and (D) of Table A1.1 were, therefore, rejected, and the question of number of shaft inputs to the main transmission (Option (A) or (B) considered. The two-shaft input system (B) was proposed for the Model 222, as indicated in Table A1.1, where separation of normal engine drive and cross-shaft systems precludes, to the maximum possible extent, one system's failure affecting the other. It is also heavier and more complex than Option (A). As to relative safety, the concept of Option (A) has been consistently considered safe enough for application to tandem rotor helicopters. Option A was thus selected.

Various relationships of wing tip and major nacelle elements - tilting rotor nacelle, fixed tail-fairing, and fixed engine nacelle - were considered as shown in Figure A1.8. Configuration (1) with engine outboard and aft of the rotor nacelle was selected as the only case providing feasible component mounting and drive possibilities.

The nacelle drive system of the selected arrangement requires three gearboxes - an engine transmission gearing engine power to the XV-15 cross-shaft, an intermediate transmission gearing the rotor to the cross-shaft, and a main (rotor) transmission as a speed reducer to rotor rpm. The first two transmissions are clearly bevel gearboxes with the main transmission including a single stage planetary system with the main transmission including a single stage planetary system to handle the rotor torque increase with symmetrical gear loading. The XV-15 cross-shaft speed at hover is 6,392 rpm and it was felt this should be held for conservatism and minimum XV-15 modification, although a higher speed is desired from the engine gearbox viewpoint. The engine transmission gear ratio is thus 3.11:1 using the T53 direct drive engine hover input rpm of 19,846. The Boeing Vertol 26-foot diameter rotor hover speed is held to the Model 222 aircraft value of 551 rpm making the cross-shaft-to-rotor reduction ratio requirement about 11.6:1. It was felt initially that to keep torque (and thus weight) low right up to the rotor, just about all the reduction should be taken in a main transmission having a spur mesh and then a single planetary stage while holding the intermediate bevel box near 1:1 ratio. Table A1.2 summarizes three options using this approach - $(A)_1$, $(A)_2$, and E. The schematics and sketch layouts of these are shown in Figures A1.9 through A1.17. The primary variable is the configuration of the main transmission input spur gear set. The first option had the input shaft and pinion offset vertically (with nacelle horizontal), thus displacing the rotor thrust off alignment with the tilt axis by an amount equal to the gear mesh canter distance, thereby increasing the load on the XV-15 tilt actuators. Since this is undesirable, the option was dropped. The second option was to place the input pinion in a laterally offset position. Here the input shaft position was such that the intermediate gearbox moved too far laterally on the cross-shaft and interfered with a nacelle mounting trunnion The option was rejected. The third Option (E) involved an attempt to reduce lateral offset, and thus the interference noted, by using an internal spur gear mesh. It appears generally feasible but the internal spur system is not desirable.

It was recognized that the 11.6:1 ratio required between cross-shaft and rotor could also be achieved by upping the intermediate transmission reduction ratio and employing only a single planetary stage in the main transmission, thus letting the input shaft to the latter come in right on center. This arrangement was defined as Option (F). It has four less gears, a higher efficiency, and probably weighs less than (E) even though the main transmission input shaft is carrying higher torque. A potential problem was that since the input shaft was set on center, a location for rotor instrumentation electrical transfer sliprings was not obvious. With the offset input shaft configuration such transfer could be made.

A rotor driven central shaft was carried right back to the aft face of the main transmission to provide a basis for electrical power and signal transfer to equipment in the aircraft - this is not possible to Option (F). The possibility of telemetering strain gage signals from the rotor to a fixed point in the aircraft was reviewed. This scheme is apparently feasible assuming a transmitter can be mounted on the hub forward face and input electric power can be transferred through the rotational barrier (a transfer ring looks possible within the upper controls assembly, aft of the hub). The Boeing Vertol laboratory has successfully used telemetering equipment of the type required. Based on this information, Option (F) was considered viable and selected as baseline over (E). The reduction ratios of main and intermediate transmissions can be optimized within the overall requirement to both minimize planetary ring gear diameter and the size, within nacelle trunnions, of the intermediate box. For preliminary size estimates, the ratio of the main box was assumed as 5.2:1 (same as YUH-61A), resulting in a 2.23:1 ratio for the intermediate box.

CONFIGURATION OPTION*	A	B	C	D
SIMILAR TO:	---	B/V M-222	BELL XV-15	---
ENGINE ARRANGEMENT	FIXED	FIXED	TILTING	TILTING
ENGINE-ROTOR TRANSMISSION DRIVE	BEVELS	BEVELS	SPURS	BEVELS
NO. OF ROTOR TRANSMISSION INPUTS	1	2	2	2
NO. OF GEARBOXES PER AIRCRAFT	7	7-9**	5	7
NO. OF GEARS PER AIRCRAFT	26	32	31	32
ESTIMATED EFFICIENCY, 2-ENGINE, η_T	97.7	97.7	97.7	---
ESTIMATED WEIGHT, LB.	1,395	1,688	1,469	---
NACELLE MOUNTING OPTIONS AVAILABLE	STRADDLE OR OVER-HUNG	STRADDLE OR OVER-HUNG	OVERHUNG ONLY	OVERHUNG ONLY

*SEE SKETCHES
**PER MODEL 222 DRAWINGS

TABLE A1.1. TILT ROTOR DRIVE SYSTEM CONFIGURATION OPTIONS

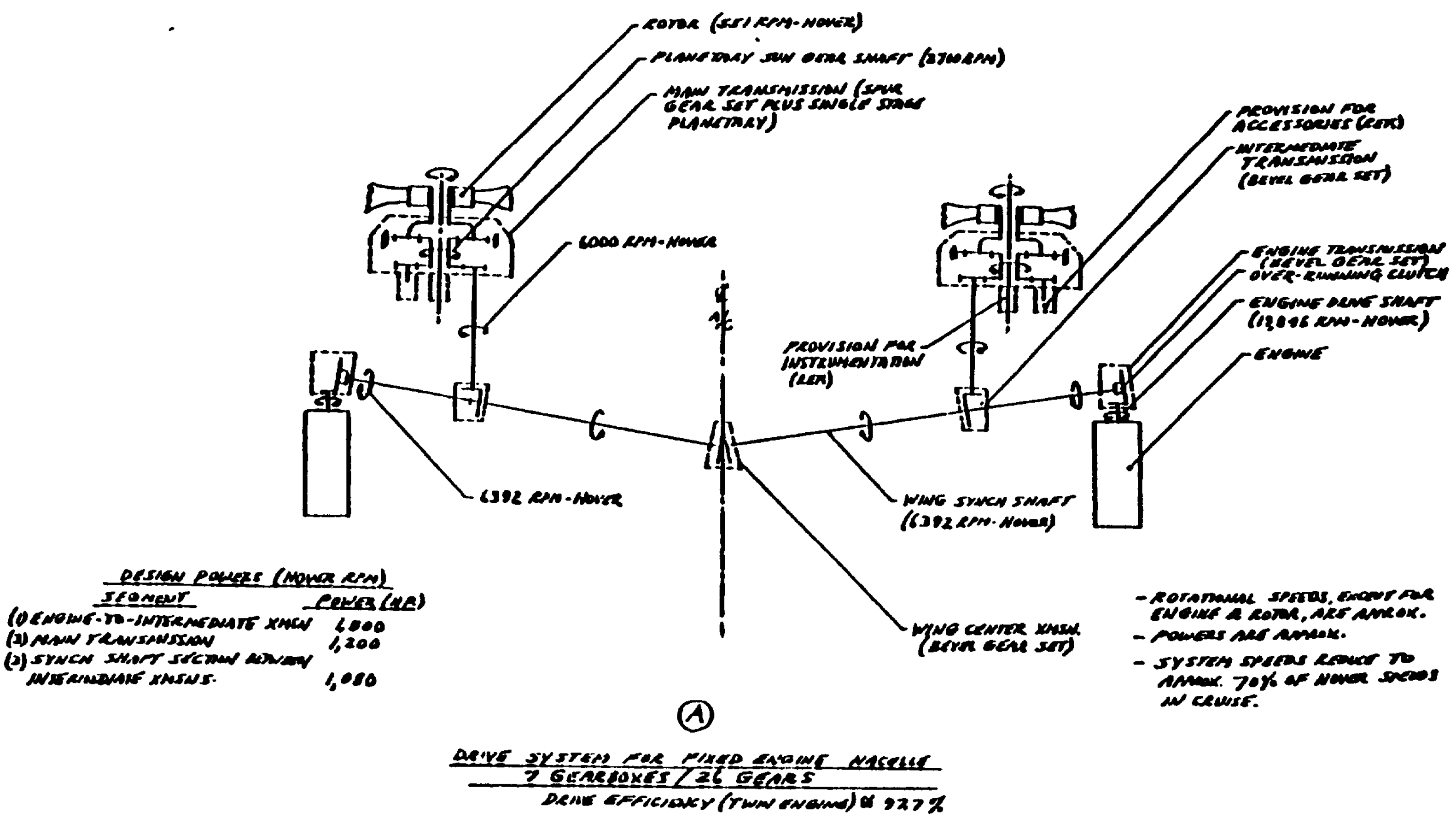


Figure A1.1. DRIVE SYSTEM LAYOUT FOR FIXED ENGINE NACELLE DESIGN

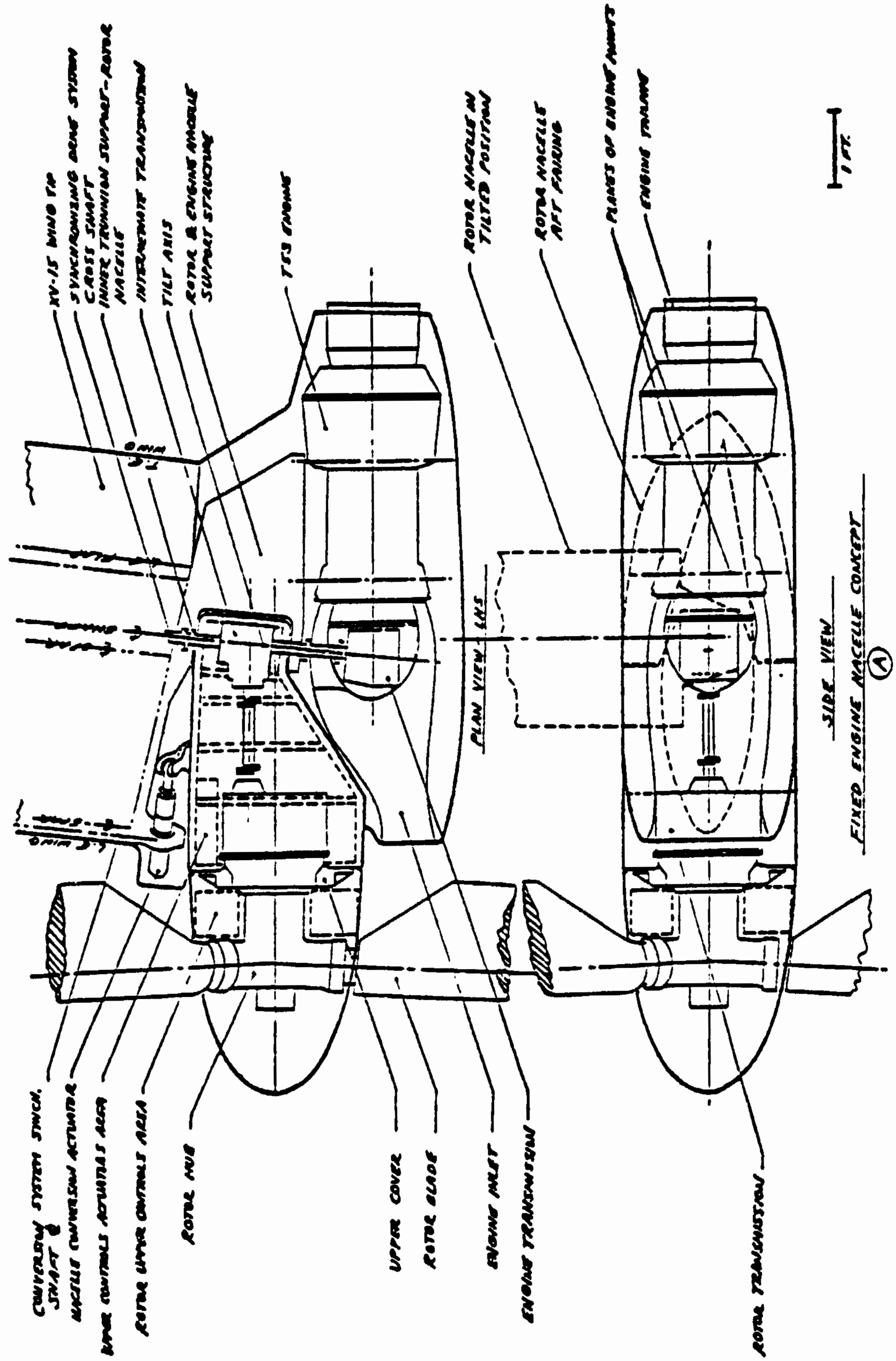


FIGURE A1.2. FIXED ENGINE, TILT ROTOR NACELLE DESIGN CONCEPT

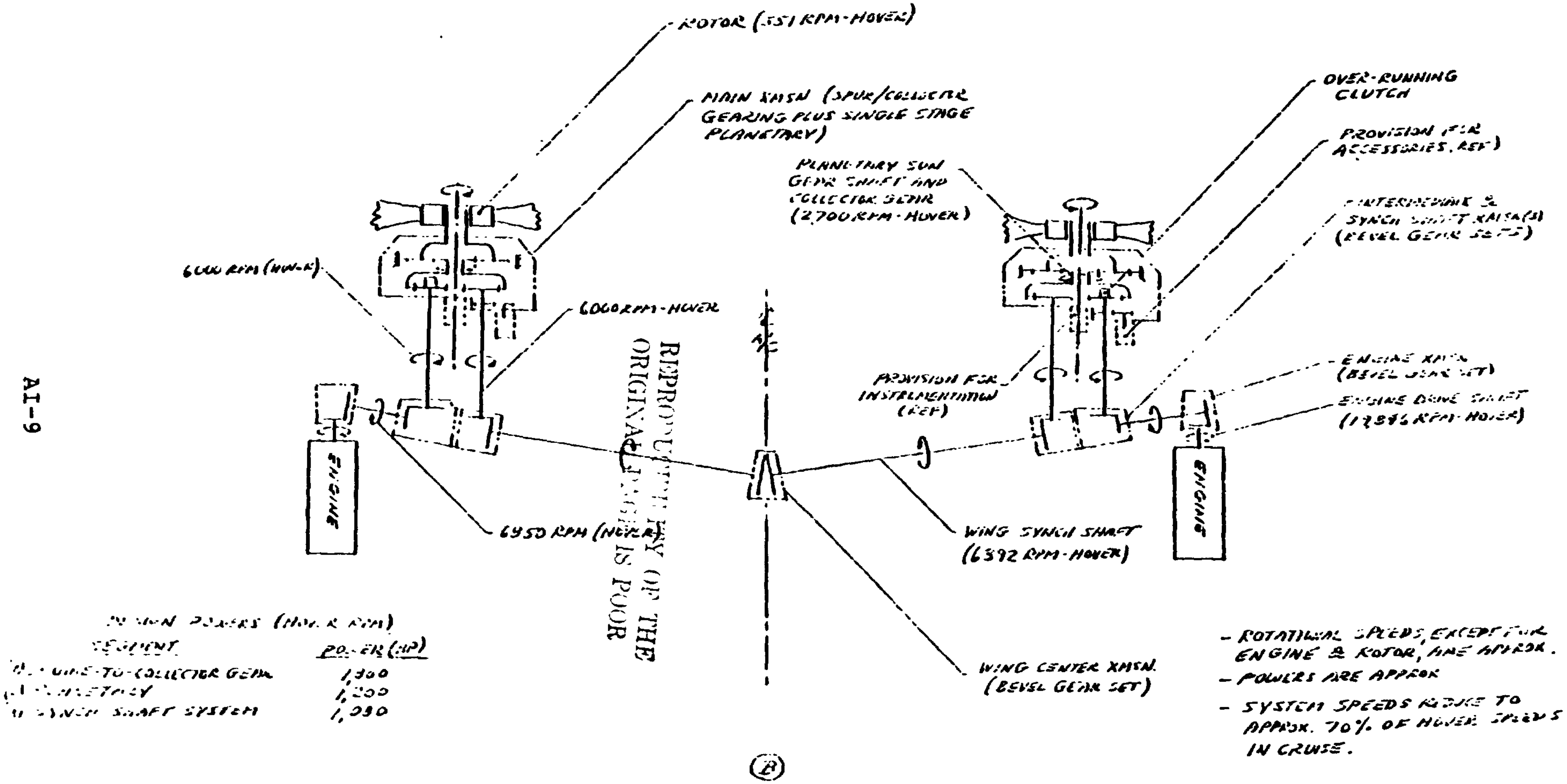


FIGURE A1.3. DRIVE SYSTEM FIXED ENGINE NACELLE - 7 GEARBOXES/32 GEARS - DRIVE EFFICIENCY (TWIN ENGINE) ≅ 97.7%

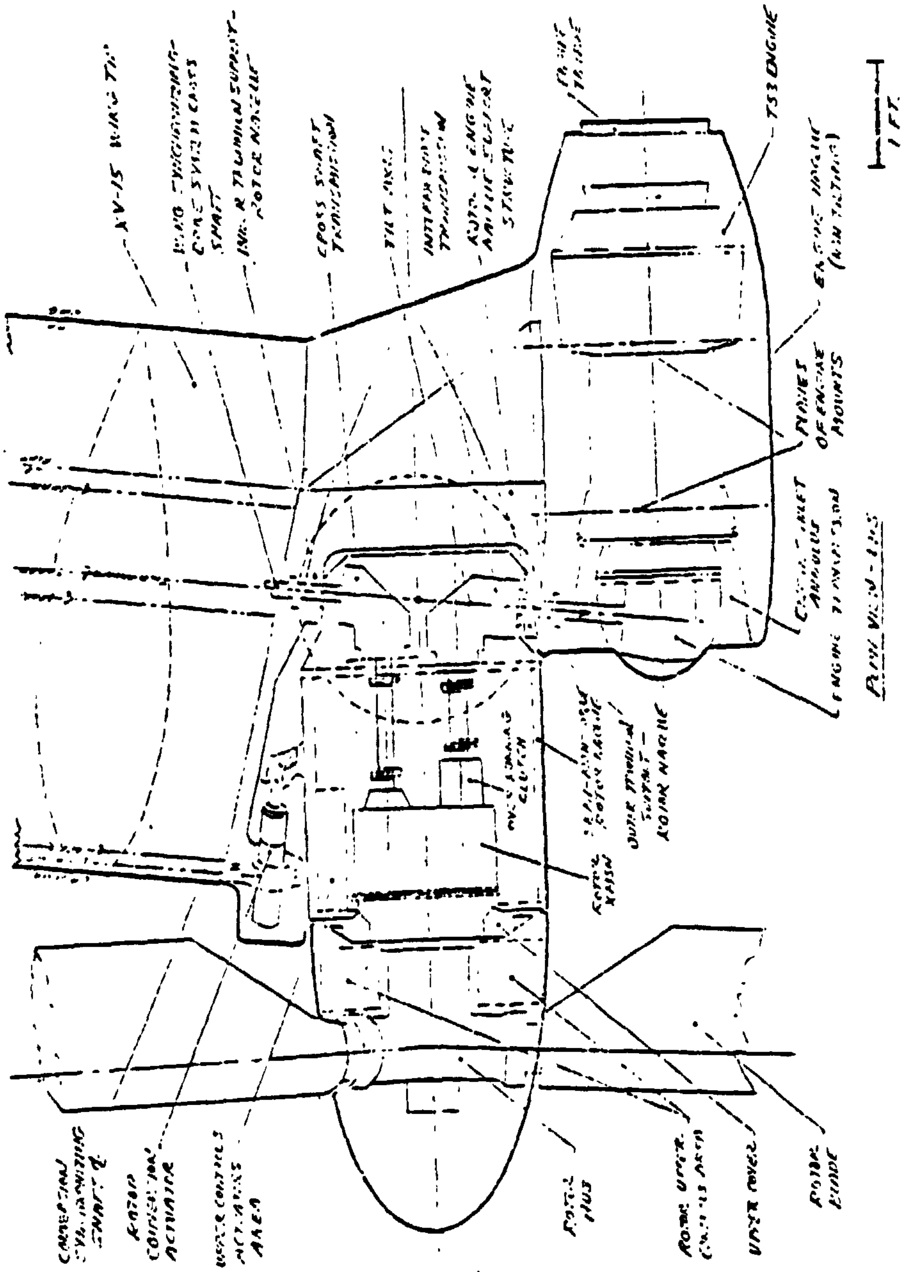


FIGURE A1.4. BASELINE FIXED ENGINE NACELLE CONCEPT

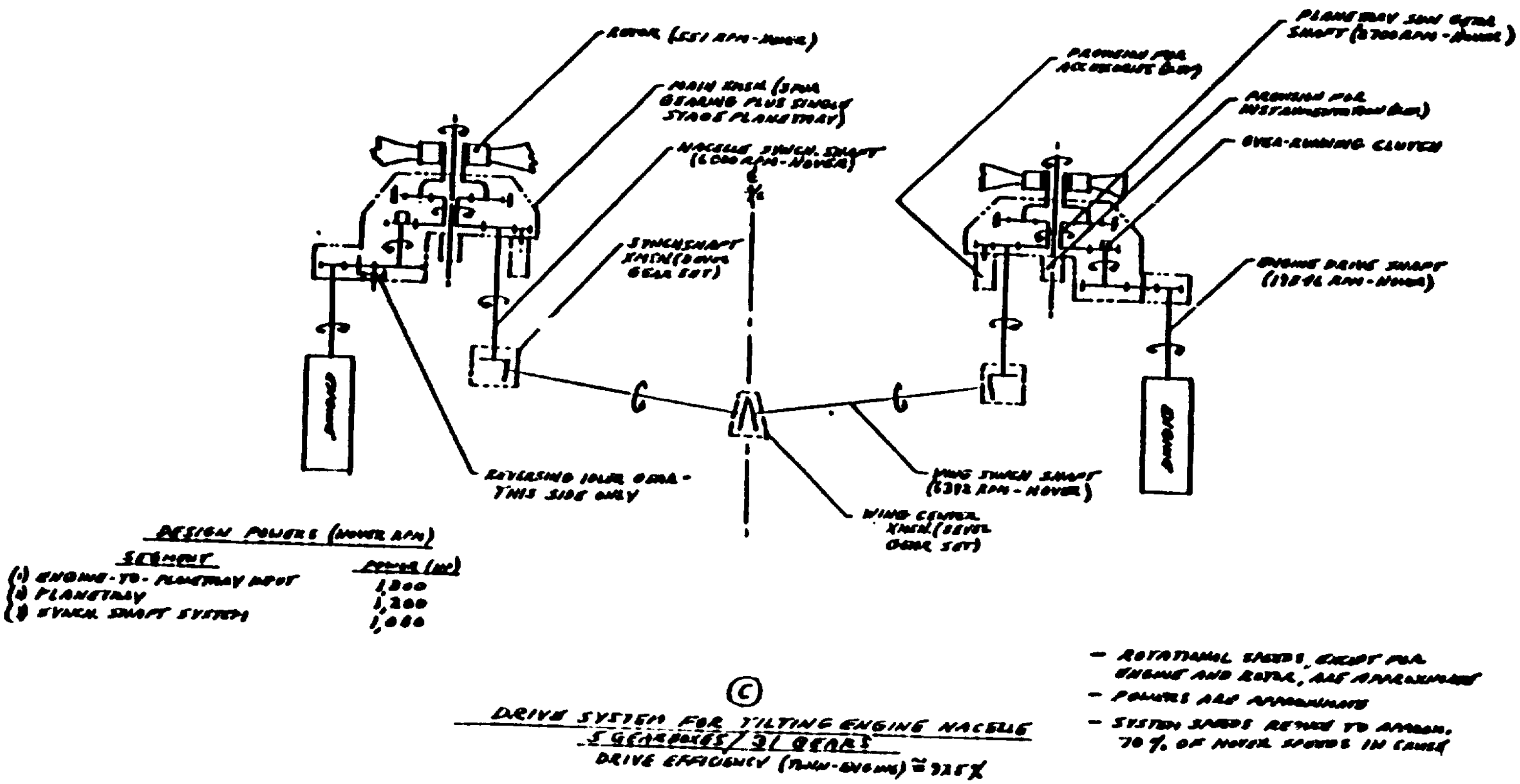


FIGURE A1.5. DRIVE SYSTEM ARRANGEMENT FOR TILTING ENGINE NACELLE DES: N

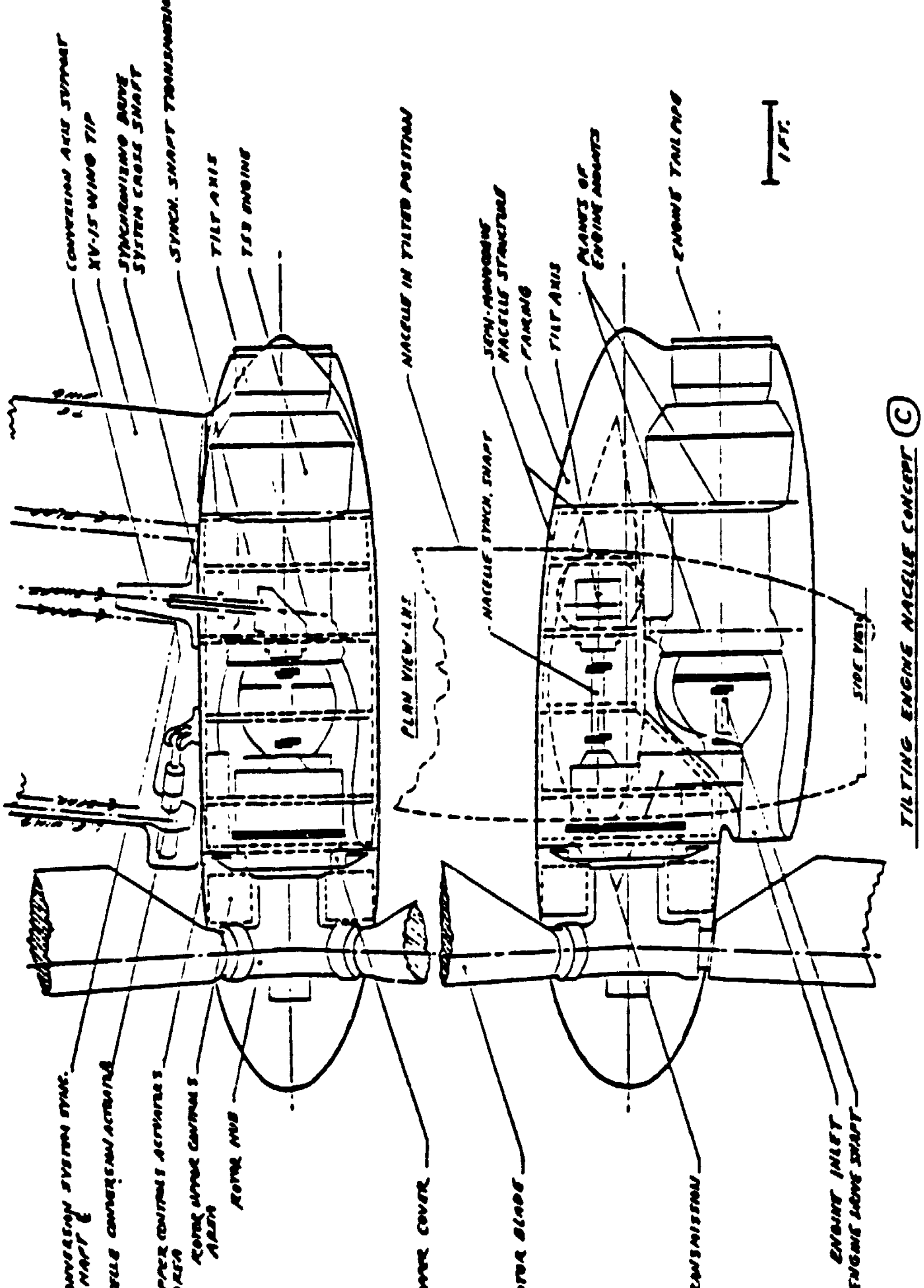


FIGURE A1.6. TILTING ENGINE NACELLE DESIGN CONCEPT

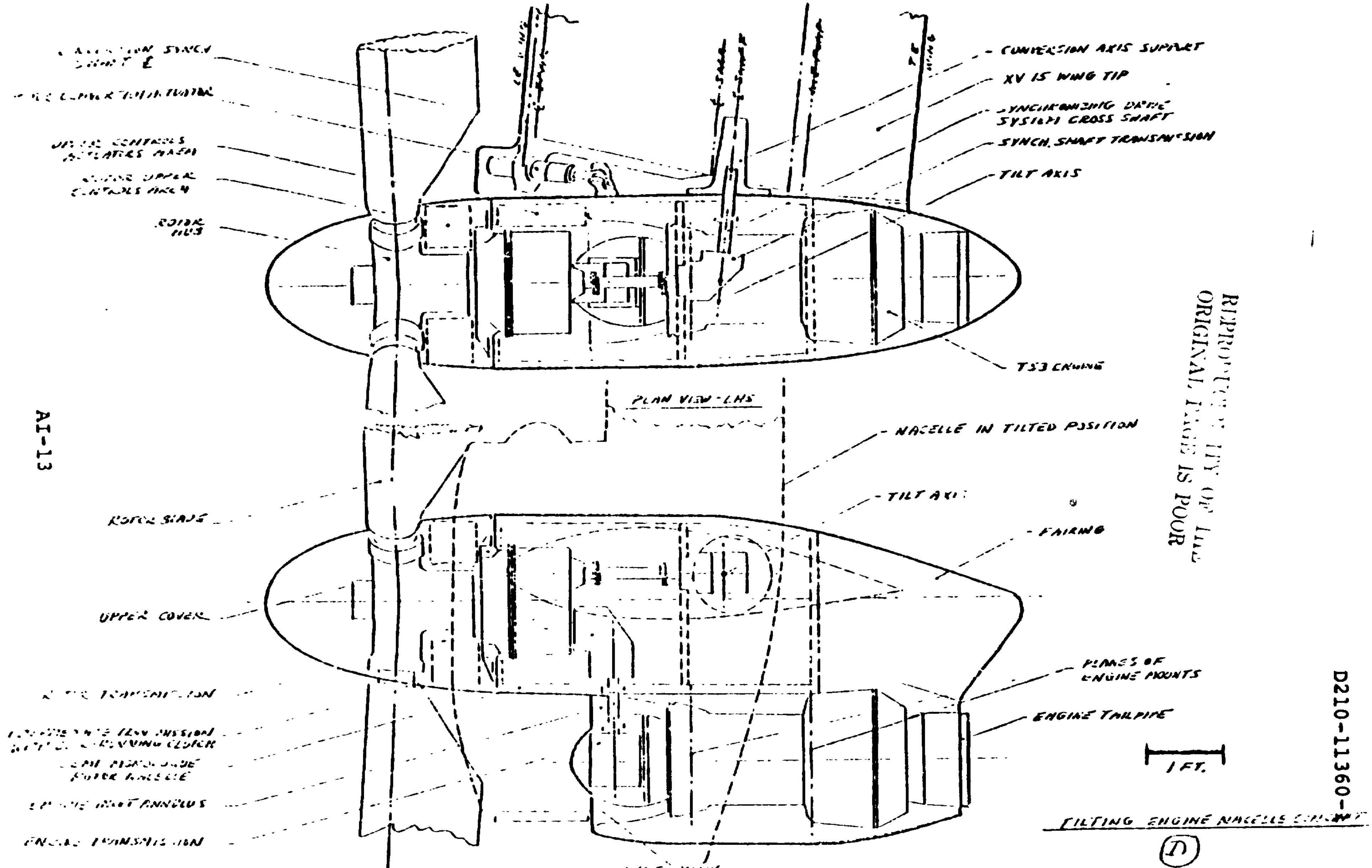
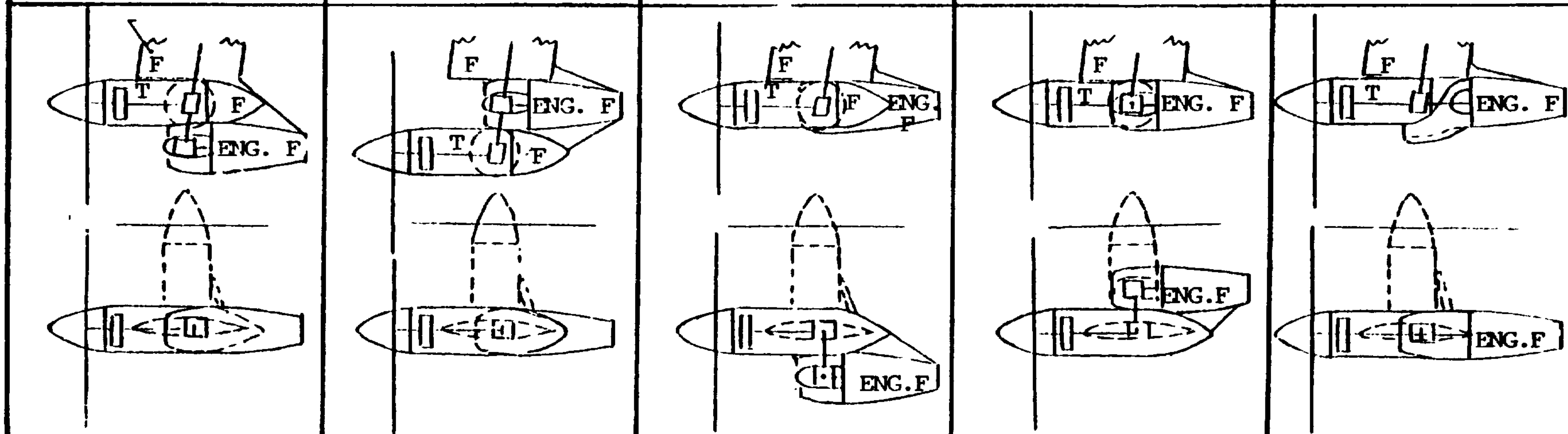


FIGURE A1.7. ALTERNATE TILTING ENGINE NACELLE CONCEPT

(1) OUTBOARD & AFT	(2) INBOARD & AFT	(3) UNDER & AFT	(4) OVER & AFT	(5) DIRECTLY AFT

PLAN VIEW

SIDE VIEW



(1) OUTBOARD & AFT	(2) INBOARD & AFT	(3) UNDER & AFT	(4) OVER & AFT	(5) DIRECTLY AFT
o OPTION SELECTED AS MOST FEASIBLE F - FIXED T - TILTING	o REJECTED - ROTOR NACELLE MOUNTING AND TILT DRIVE SYSTEMS LONG AND HEAVY, AND ENGINE INTAKE IS BLOCKED BY THESE ELEMENTS	o REJECTED - GEARING CANNOT ACCOMMODATE ROTOR NACELLE TILT WITH THE ENGINE DIRECTLY IN LINE - NEEDS LATERAL OFFSET AS IN (1) o ADDITIONAL CFARBOX BEYOND SYSTEM (1) DUE TO LOW ENGINE	o REJECTED AS IMPRACTICAL - ENGINE AND TILTED ROTOR NACELLE INTERFERE	o REJECTED - GEARING CANNOT ACCOMMODATE ROTOR NACELLE TILT WITH ENGINE DIRECTLY IN LINE - NEEDS LATERAL OFFSET AS IN (1) o ENGINE MUST BE LOCATED FAR AFT TO ALLOW INLET SYSTEM; MOUNTING IS DIFFICULT

FIGURE A1.8. FIXED ENGINE NACELLE LOCATIONS WITH RESPECT TO TILTING ROTOR NACELLE

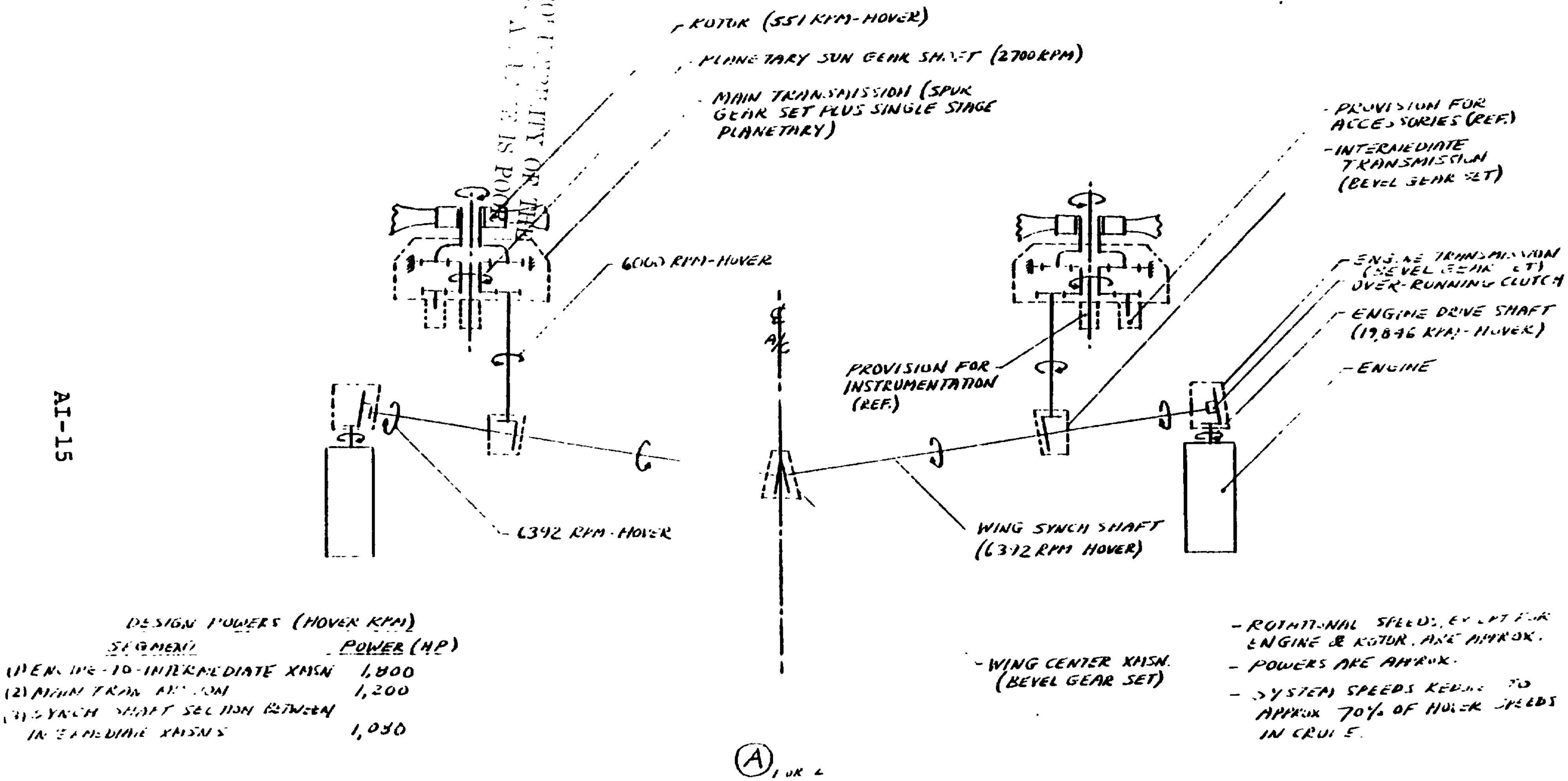


FIGURE A1.9. DRIVE SYSTEM FOR FIXED ENGINE NACELLE - 7 GEARBOXES/26 GEARS - DRIVE EFFICIENCY (TWIN ENGINE) ≅ 97.7%

TABLE A1.2

COMPARISON OF DRIVE SYSTEM CONCEPTS WITH FIXED ENGINES - B/V ROTOR/NACELLE ON NASA XV-15 TILT ROTOR

(NOTE - ALL CONCEPTS HAVE SINGLE INPUT SHAFTS TO MAIN XMSN.)

OPTION	A₁	A₂	E*	F
MAIN XMSN. DESCRIPTION (SEE SCHEMATIC DWGS)	EXTERNAL SPUR SET PLUS SSP (INPUT PINION AT T'P OR BOTTOM OF CENTRAL SPUR GEAR)	EXTERNAL SPUR SET PLUS SSP (INPUT PINION INBOARD OR OUTBOARD OF CENTRAL SPUR GEAR)	INTERNAL SPUR SET PLUS SSP (HORIZ. ALIGN. OF SPUR SET)	SSP ONLY
NUMBER OF GEARBOXES	7	7	7	7
NUMBER OF GEARS (POWER)	26	26	26	22
EST. η** (A.E.O.)	97.7%	97.7%	97.7%	98.2%
EST. WEIGHT (LB)	1395	1395	>1395	<1395
HANDING				
- ENGINE XMSNS	YES	YES	YES	YES
- INTERMEDIATE XMSNS	NO - LH & RH BUILD-UP	NO - LH & RH BUILD-UP	NO - LH & RH BUILD-UP	NO - LH & RH BUILD-UP
- MAIN XMSNS	NO	NO - LH & RH BUILD-UP	NO - LH & RH BUILD-UP	NO
INTERMED. XMSN LATERAL LOCATION PROBLEMS	NO	YES - INTERFERES WITH TILT NACELLE STRUCT. MT'G TRUNNION	NO	NO
XV-15 TILT ACTUATOR LOADING PROBLEMS	YES - INCREASES ACTUATOR LOADING	NO	NO	NO
PROVISIONS FOR INSTRUMENTATION SLIP RINGS	YES	YES	YES	NONE OBVIOUS - (USE TELEMETERING DIRECT FROM ROTOR ?)

NOTES OPTION A WAS SELECTED PREVIOUSLY OVER OPTIONS B, C, AND D. OPTION B WAS SIMILAR TO THAT OF THE B/V M-222 WITH TWO INPUT SHAFTS TO MAIN XMSN. OPTIONS C AND D EMPLOYED A TILTING ENGINE CONCEPT.

* SIMILAR TO PROPOSED B/V M222

** ASSUMING $\eta_{SPUR\ MESH} = 99.5\%$; $\eta_{BEVEL\ MESH} = 99.5\%$

$\eta_{SSP} = 99.25\%$

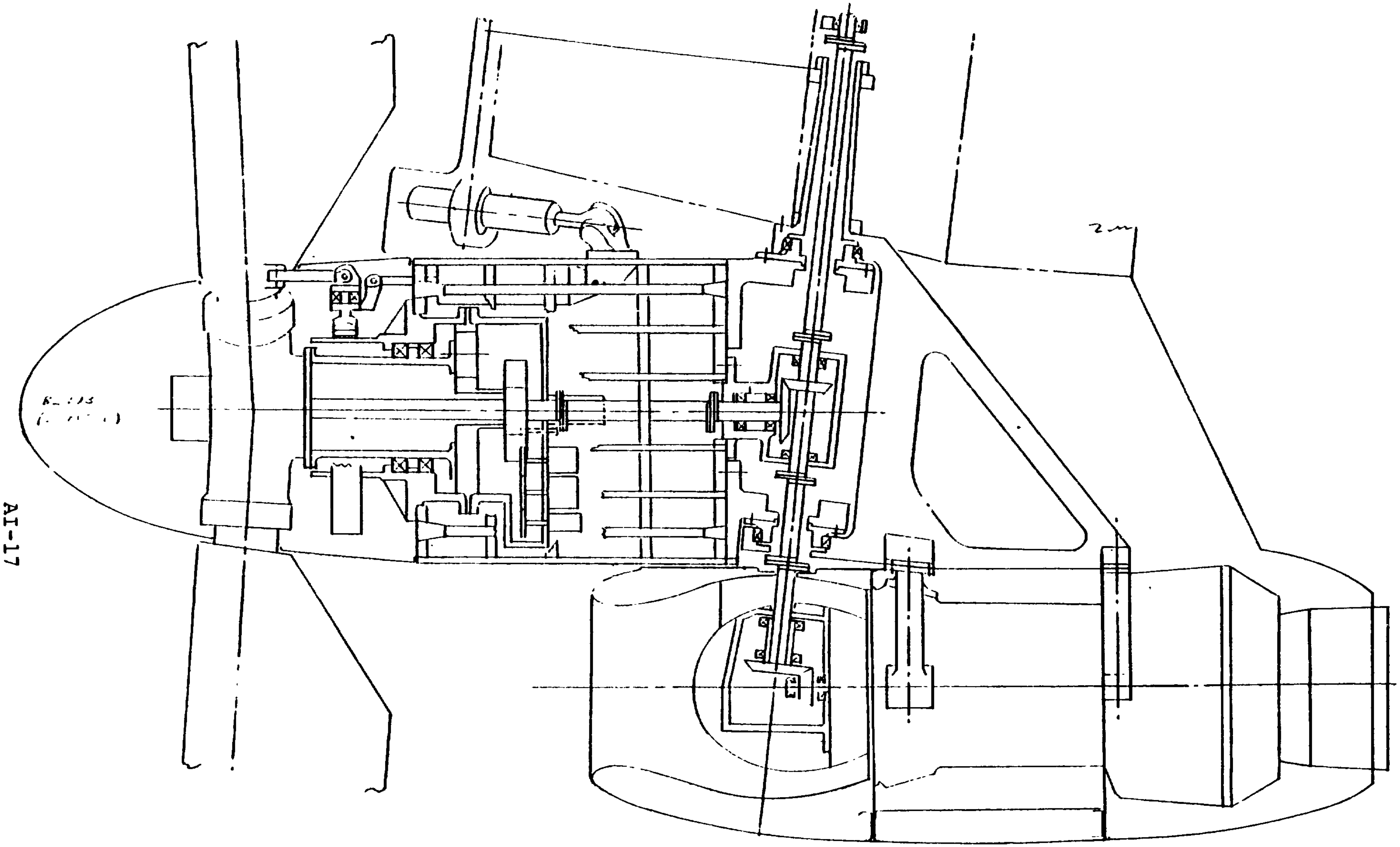


FIGURE A1.10. FIXED ENGINE NACELLE WITH BOEING VERTOL ROTOR FOR XV-15 AIRCRAFT VARIANT WITH MAIN TRANSMISSION INPUT PINION HIGH FOR INTER-CHANGEABLE MAIN TRANSMISSIONS, LHS-RHS, ROTOR AT BL 193 - PLAN VIEW

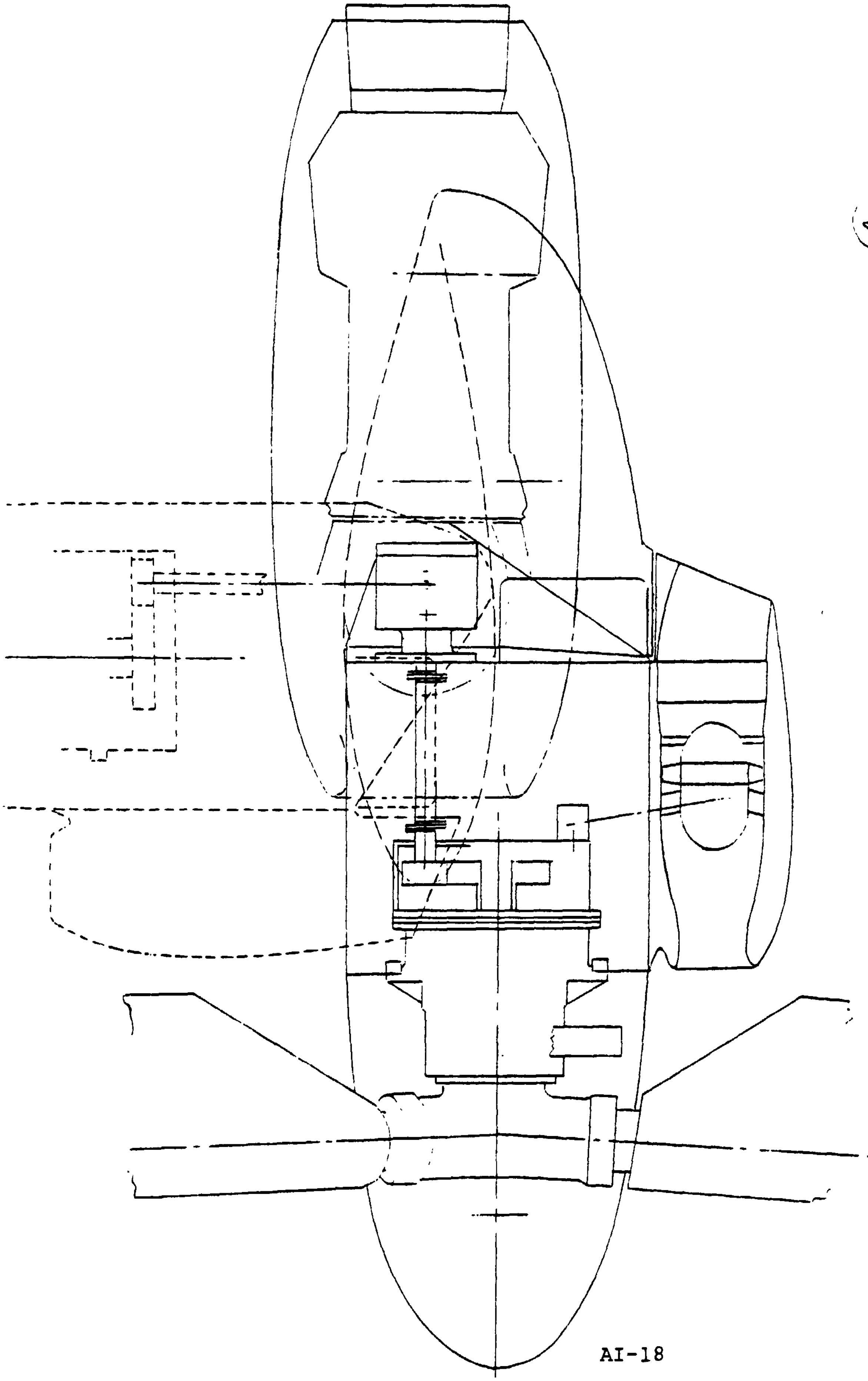


FIGURE A1.11. SIDE VIEW OF FIGURE A1.10

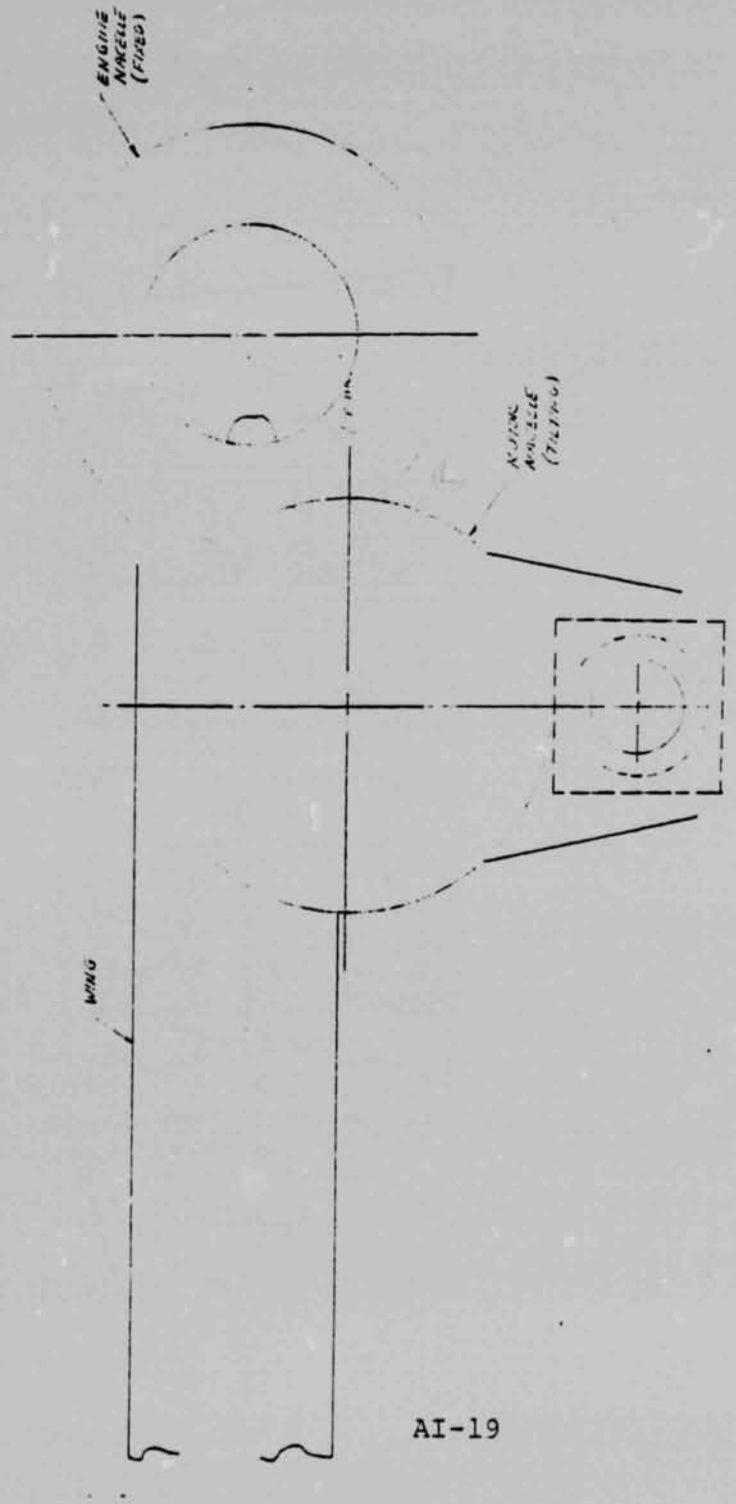


FIGURE A1.12. FRONT VIEW OF FIGURE A1.10

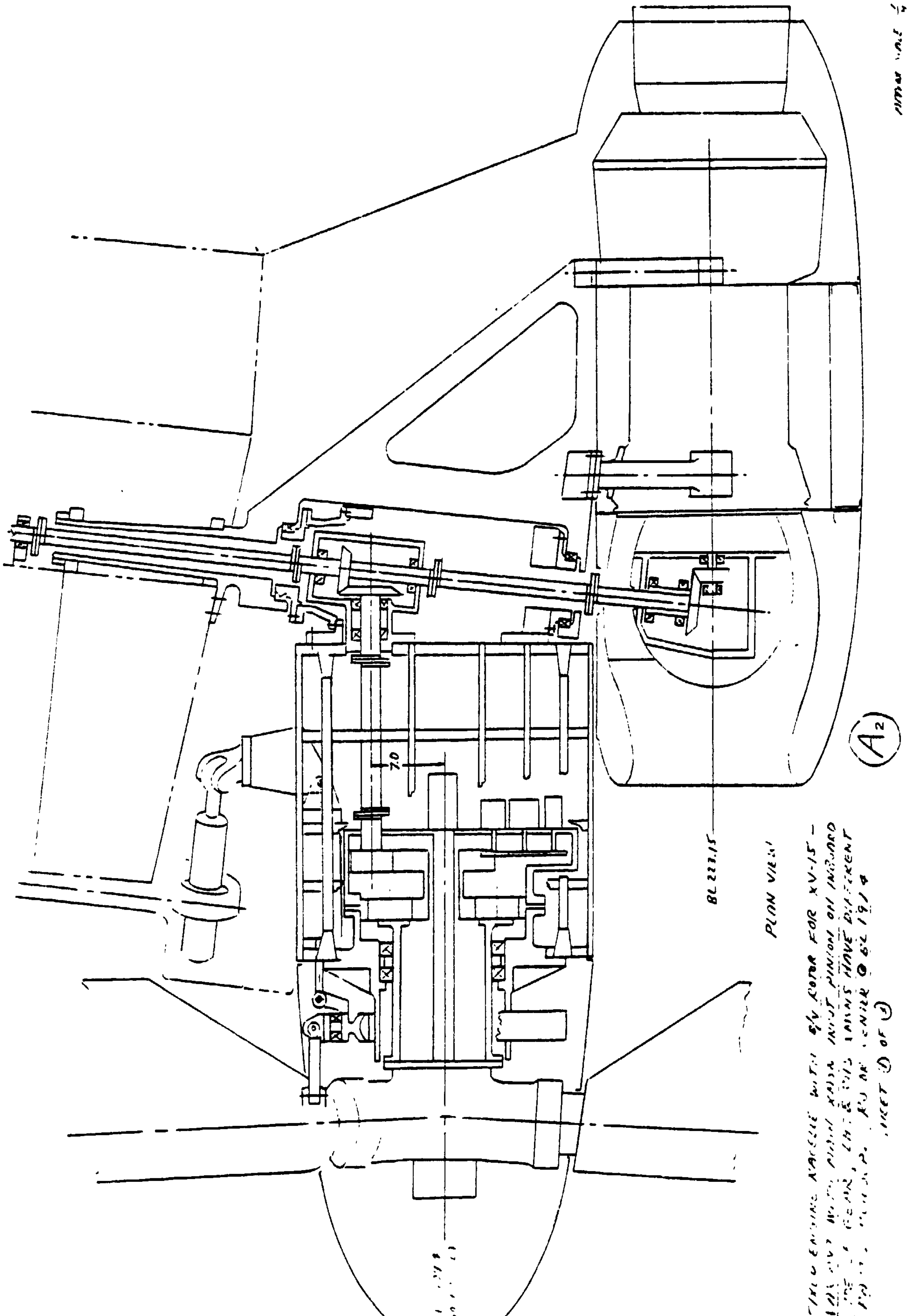


FIGURE A1.13

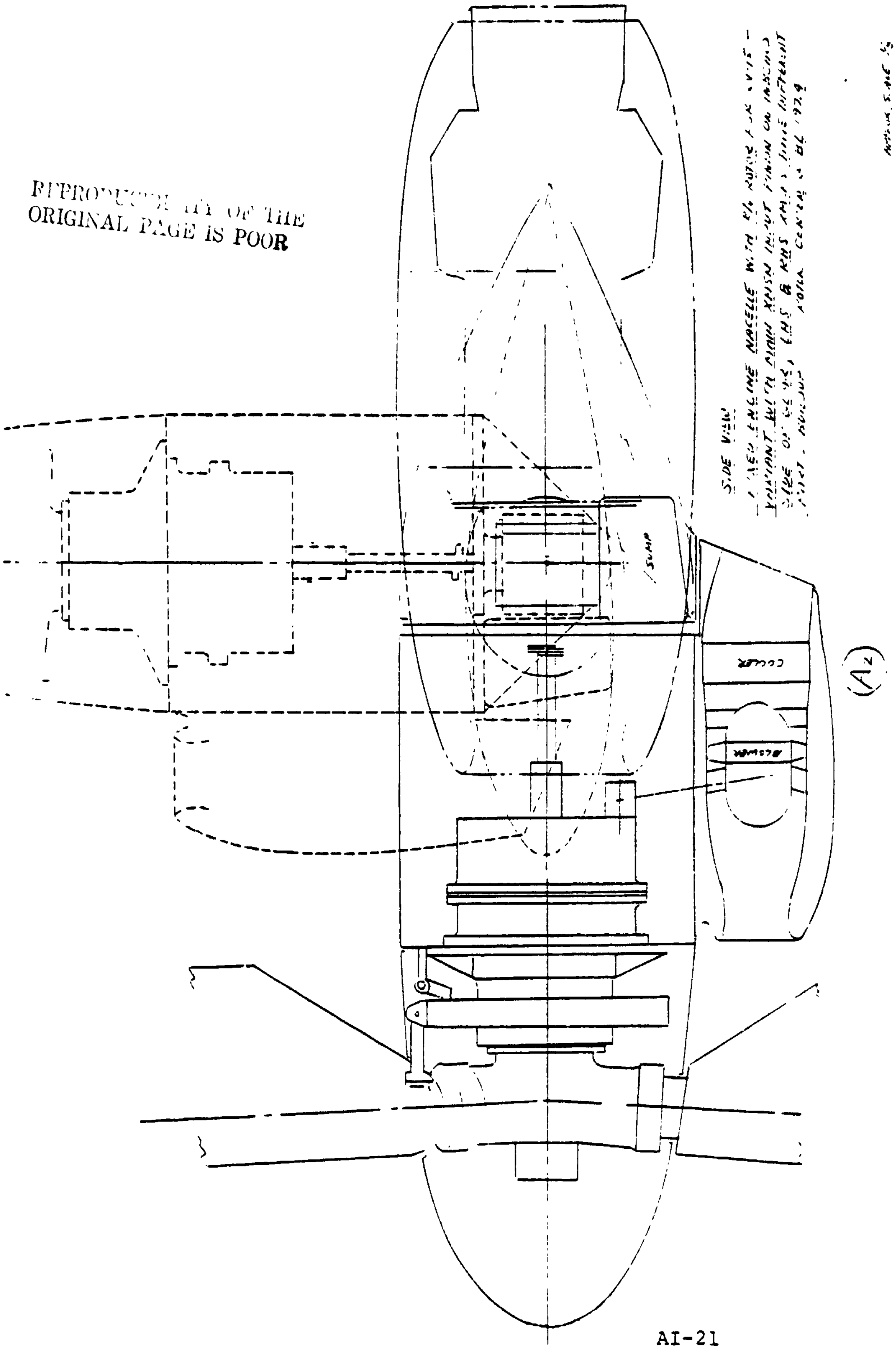
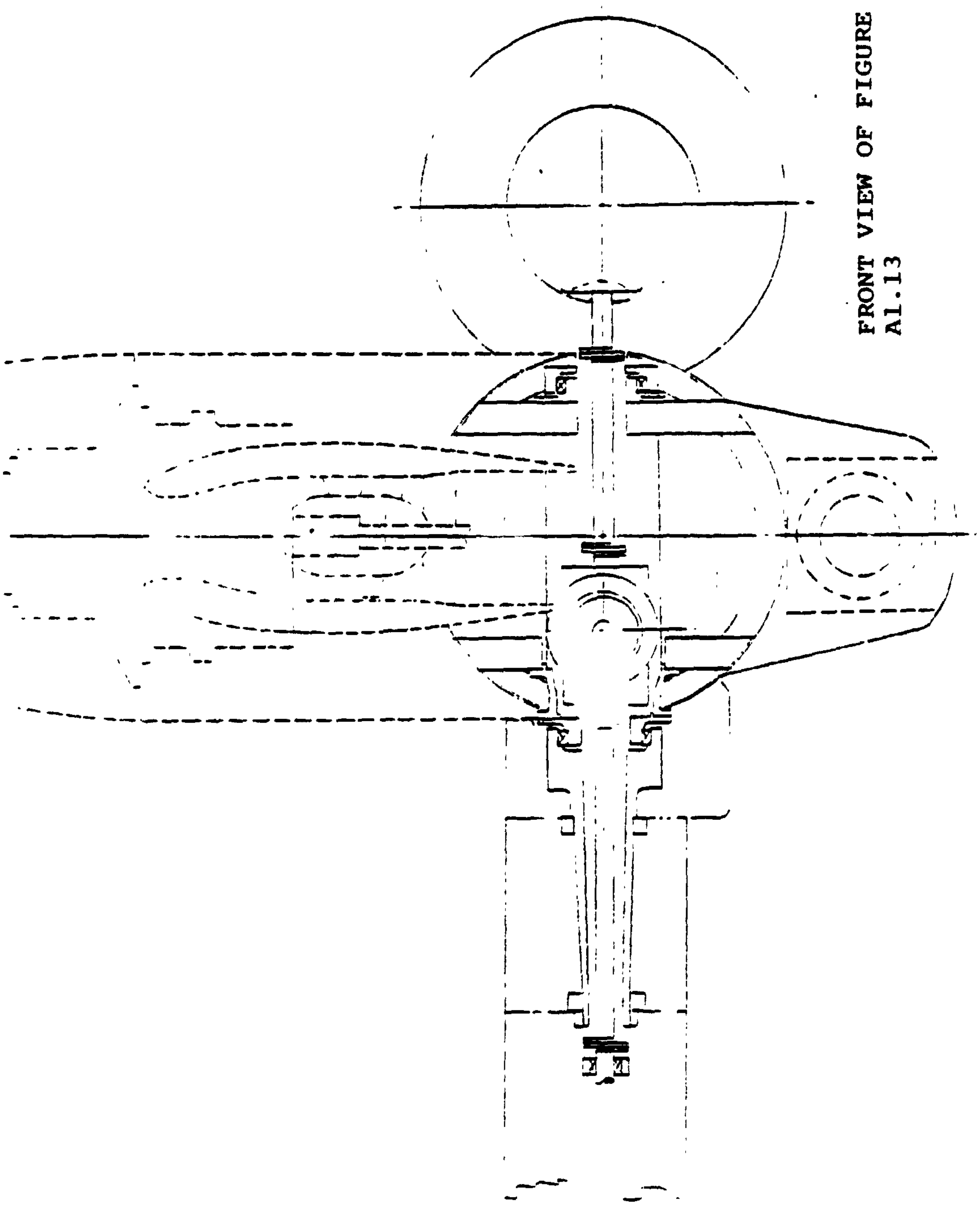


FIGURE A1.14



FRONT VIEW OF FIGURE A1.13

A_2

FIGURE A1.15

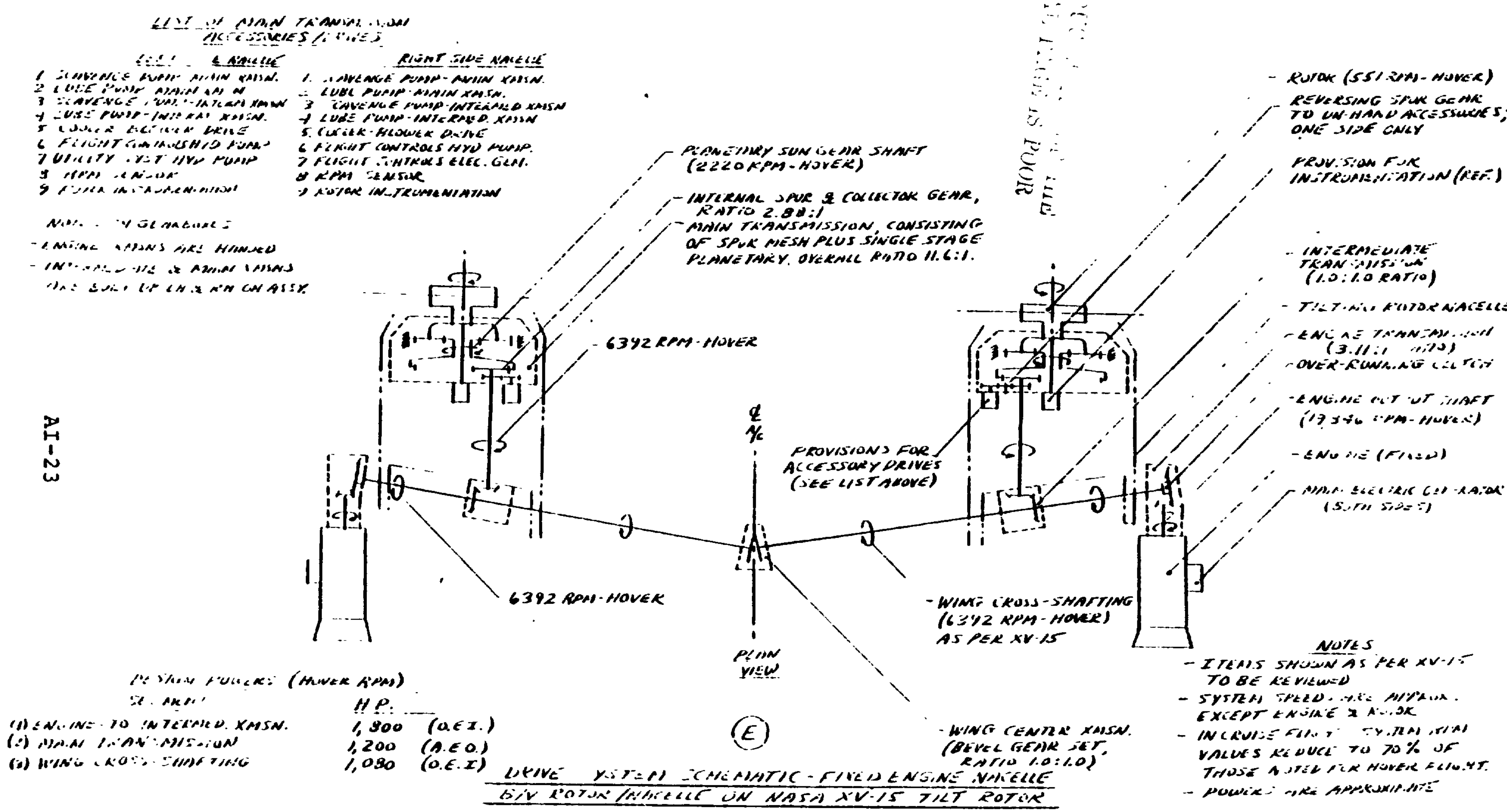


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FIGURE A1.16

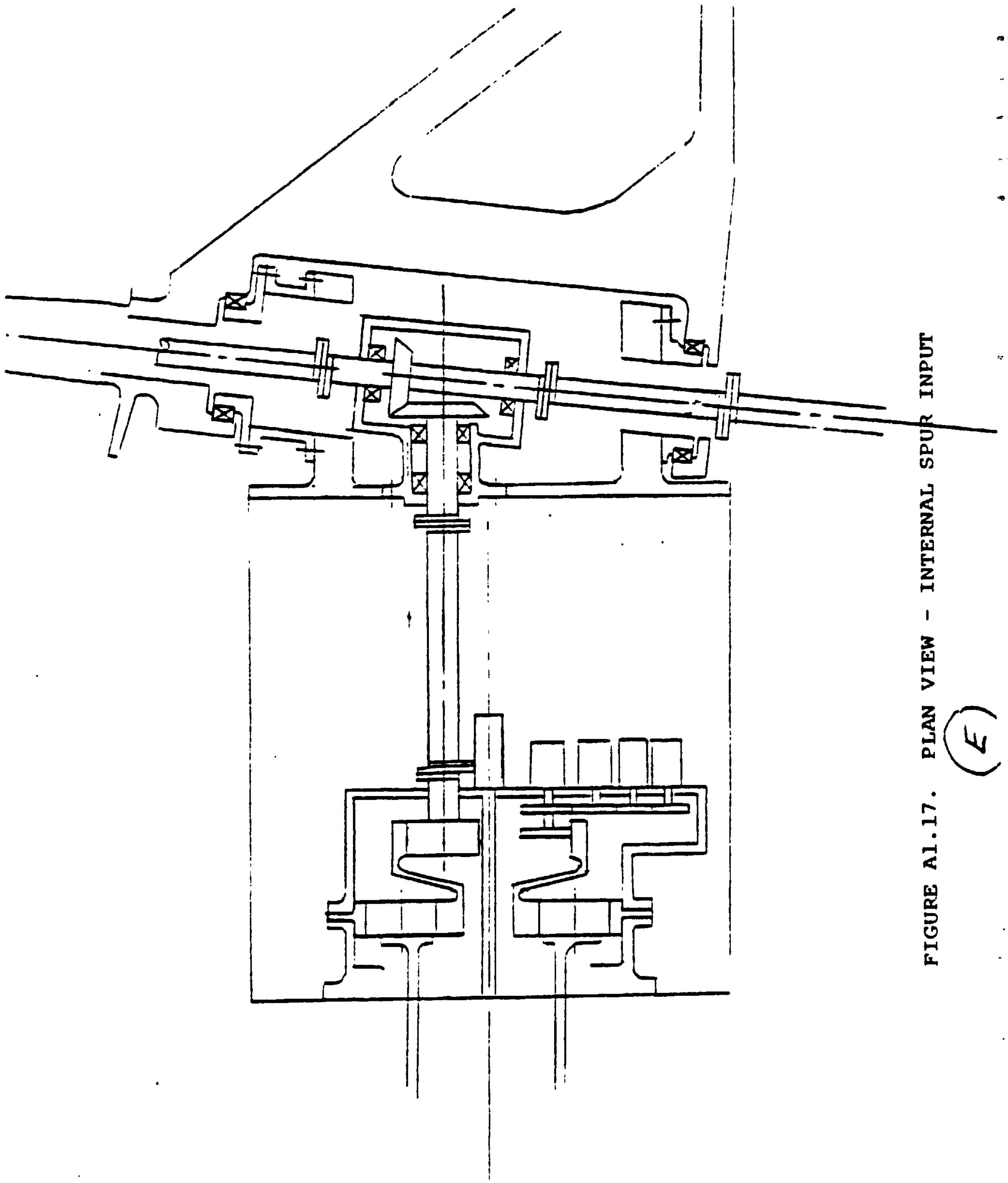


FIGURE AI.17. PLAN VIEW - INTERNAL SPUR INPUT

E

AI-2 - TILT ACTUATION SYSTEM

The installation of a Boeing Vertol rotor and nacelle on the XV-15 tilt rotor aircraft was to use the current XV-15 conversion (nacelle tilt) system. The spanwise location of the XV-15 tilt actuator for the Boeing Vertol system requires definition since our rotor is 26-foot diameter versus 25-foot for the XV-15.

A) XV-15 Conversion System Definition

The following characteristics are known:

- Nacelle angular range - 95°
- Actuator total stroke - 28.5"
- Time for full stroke - 11-14 seconds
- Actuators - mechanical ball screw jack
- Power drive - hydraulic
- Power control - electric
- Specification - Bell No. 301-947-011
- Load, normal (each) 2400 pounds extend; 3700 pounds retract
- Load, maximum operating (each) 3300 pounds extend; 5000 pounds retract

Figure A1.18 shows the conversion system schematic. Figure A1.19 shows the system related to the aircraft, and Figure A1.20 approximates the actuator kinematics. Data are taken from References 3 and 4. A ball screw jack used for nacelle tilt is trunnion-mounted on each wing tip and the screw driving nuts are mechanically interconnected via gearing and wing leading edge cross-shafting so as to cater for power or control failures on one side. Hydraulically powered motors, each integral with an actuator, drive the nuts through a clutch and brake, the latter holding the system with power off (since the high-efficiency ball screws are quite reversible). Power comes from the separate flight control hydraulic systems in the nacelles. Primary control of the motors is electric using solenoid valves and motor clutches. Separate DC busses are used for each side.

A separate lower rate emergency actuator drive and control system, catering to nacelle hydraulic or electric failures, is provided by a third hydraulic motor located on aircraft centerline and driving the cross-shafting through a clutch and gearbox. Control of this system is electric.

An emergency backup manual mechanical control (cockpit T-handle) of nacelle hydraulic motor valves caters to a total electrical failure mode, and can move the valves to bring the nacelles to the helicopter position only.

Other system features include nacelle tilt indicators in the cockpit (taken from sensors measuring mounting trunnion angular positions) and assymetric tilt angle detection where hydraulic flow is stopped and nacelles locked, and interlock switches on the aircraft landing gear to prevent inadvertent tilt down on the ground. An input is also provided to the flight control system via a mix box in the wing center section.

In Figure A1.20 it is interesting to note that, since the tilt actuator trunnion and the cross-shaft interconnection are on different axes, there is a slight "waggle" to the cross-shaft outer sections where they enter the actuator.

B) <u>Tradeoff Options for Spanwise Location of Rotor Tilt Nacelle and Conversion Actuator</u>

Table A1.3 lists three possibilities for the above locations, and some of the factors involved in a decision. Option 1 is shown in Figure A1.21 where the rotor butt line and tilt actuator relative location, in a proposed Boeing Vertol design, are like the XV-15, resulting in a 6-inch rotor-fuselage clearance and a short-coupled actuator output drive to the nacelle. Option 2 is shown in Figure A1.22 where (in this sample picture) the rotor is moved out 4 inches to give a 10-inch rotor-fuselage clearance, but the tilt actuator is held in the XV-15 location, resulting in a longer output drive attachment to the nacelle. Option 3 is not sketched because it would look like Figure A1.21. In this case, both tilt nacelle centerline and actuator are moved out the same distance by building out the wing-tip structure up to 6 inches more (for 12-inch maximum blade-fuselage clearance).

Table A1.3 indicates by far the least change in loads and component changes in Option 1. It shows that Option 2 involves changes on the high load output side of the tilt actuator, but none on the input side and no big wingtip change. Option 3 means no actuator output side changes, being close to the tilt nacelle, but both actuator input and wingtip build-out changes are involved.

Subsequent examination of the wingtip structure caused the displacement of the rotor centerline outboard such that the rotor tip-fuselage clearance became 12 inches. This consideration and the desire to minimize modification to the XV-15 wingtip drive the selection of Option 2.

C) Straight Versus Toed-In Engines

If the engines and engine nacelles are located on a butt line parallel to aircraft centerline and at an 84.5 degree angle to the swept forward cross shafting, the left and right engine transmissions are, handed and require uncommon spares.

Figure A1.24 shows engines toed nose-in 5.5 degrees to allow right angle gearing in engine transmissions, thus allowing the possibility of many common parts side-to-side and reducing the cost of spares in the aircraft program. It was considered that the engine inlets would not present a problem. There will be a small weight penalty in the adapter mounting the engine which must reach further out to attach.

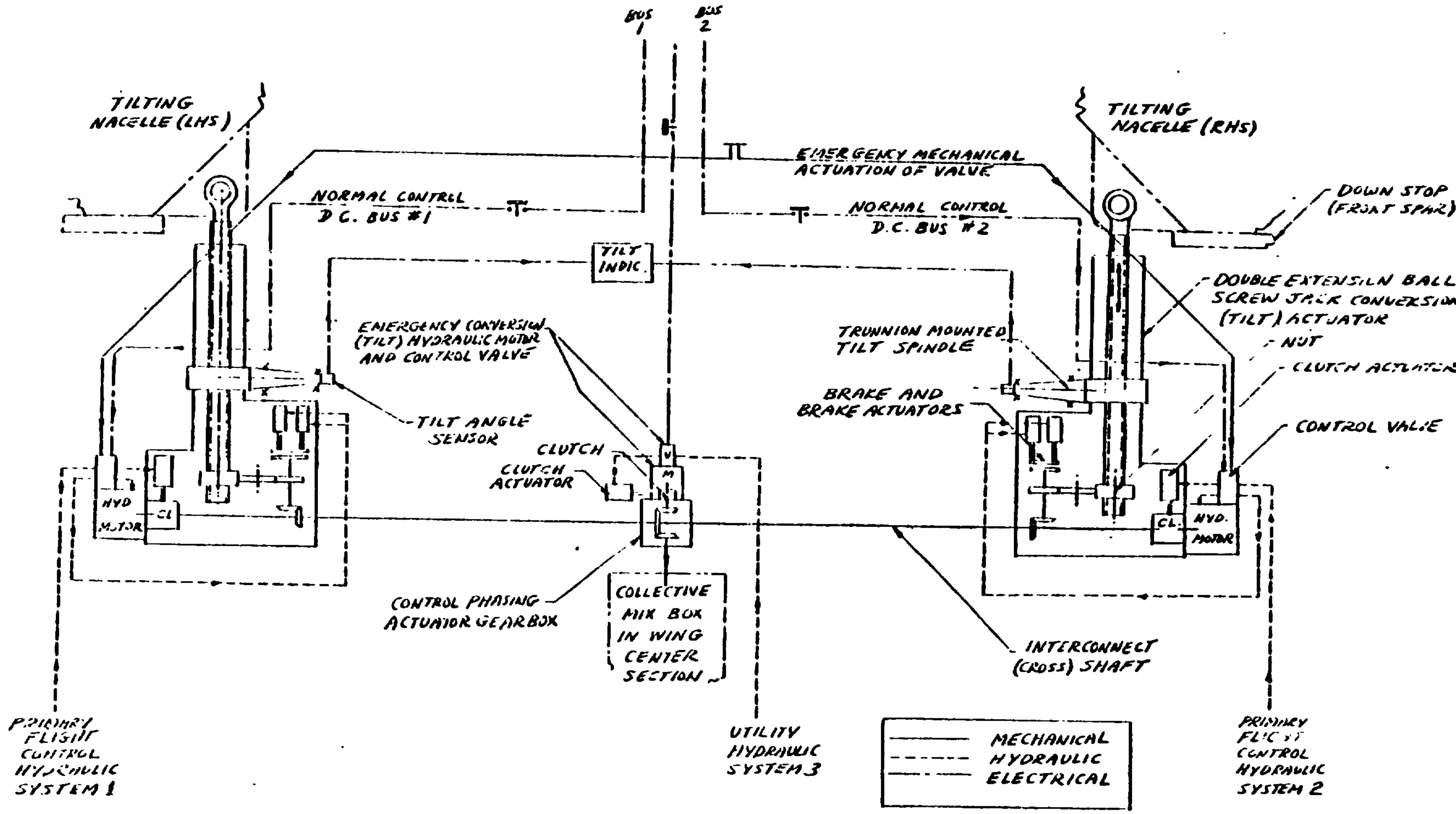
The toed-in engine arrangement has been selected as a baseline.

D) Integrated Versus "Patch-On" Engine Gearbox Arrangement

Figure A1.25 shows a comparison of two options for mounting the engine transmission on the Lycoming LTC1K-4K (T53 direct-drive) engine. Option A is closely integrated to minimize weight and overhang of the box by allowing the input pinion gear support system to intrude into engine nose space vacated by the normal T53 engine reduction gearing in the original conversion to direct drive. This arrangement is similar to that proposed for the Boeing Vertol Model 222 design as shown in layout SK222-10210 and coordinated with Lycoming at that time. The design had common lubrication for engine and Boeing Vertol gearbox, not envisioned in current Option A. The current XV-15 with the Bell gearbox does not use the scheme - see engine installation drawings. Recent review of Option A with Lycoming indicates sufficient changes in the current engine would be needed so as to require a new 60-hour PFRT at considerable cost to the envisioned HTR XV-15 modification program.

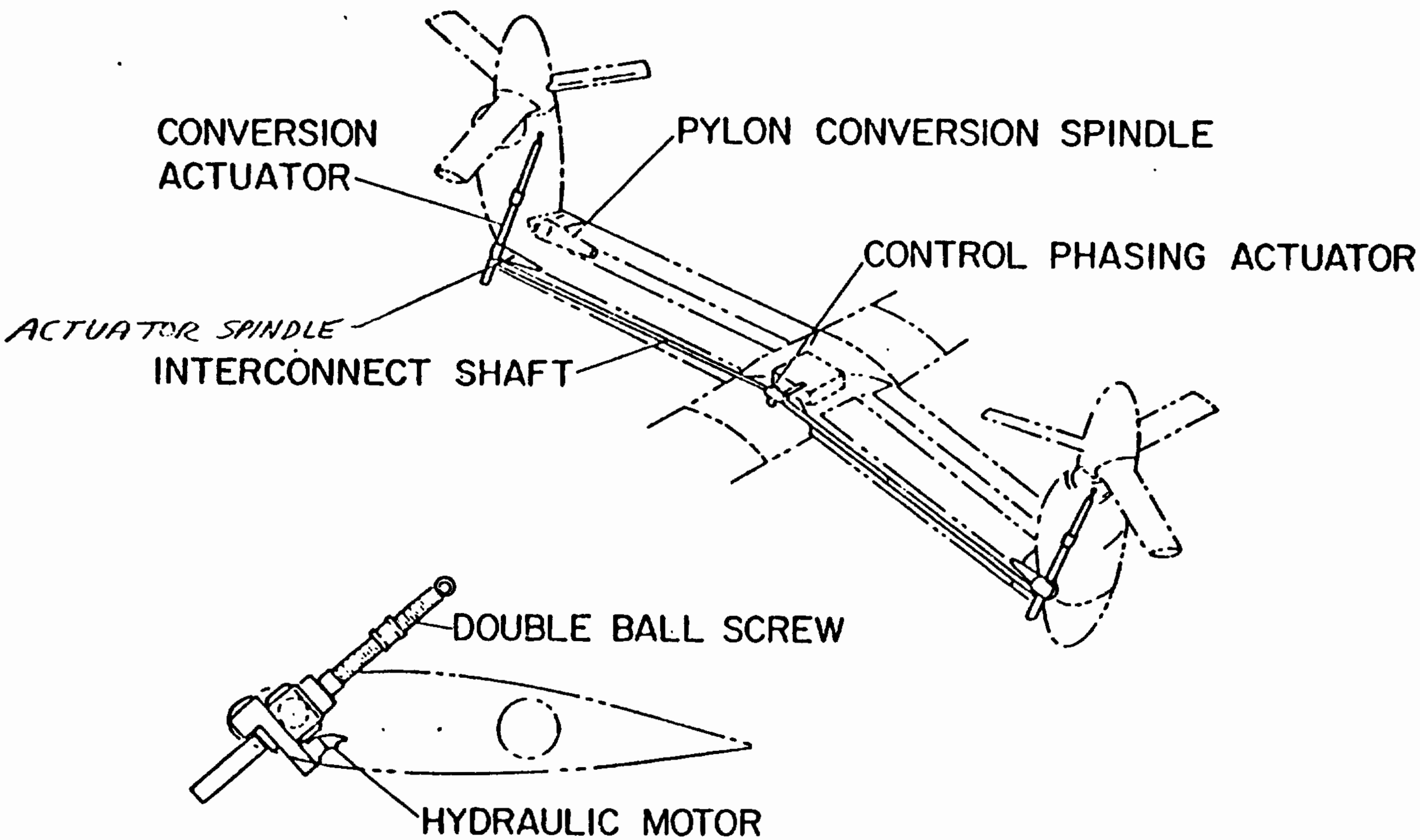
Option B of Figure A1.25 shows the nose gearbox hung or "patched" on the current engine nose with no engine revisions. The pinion gear bearing support space required is just forward of the engine nose - the pinion shaft male spline would mate with the female engine drive shaft spline and the gearbox case would be tied in at the engine mounting face stud circle. The engines currently in the XV-15 program could be used with no change and no requalification. Option B involves weight penalties over Option A - the gearbox casing is longer and overhung moments are greater; in addition, the engine is pushed aft 6.2 inches making the engine support adapter larger and heavier.

Option B, the "patch-on" gearbox, has been selected as a baseline more suitable to the aircraft modification program being studied, for reasons of reduced cost for a one of a kind installation.



SIMPLIFIED SCHEMATIC DIAGRAM -
XV-15 NACELLE CONVERSION (TILT) SYSTEM

FIGURE A1.18. SIMPLIFIED SCHEMATIC DIAGRAM - XV-15 NACELLE CONVERSION (TILT) SYSTEM



Conversion system_ XV-15 Aircraft

FIGURE A1.19. CONVERSION SYSTEM - XV-15 AIRCRAFT

2-47

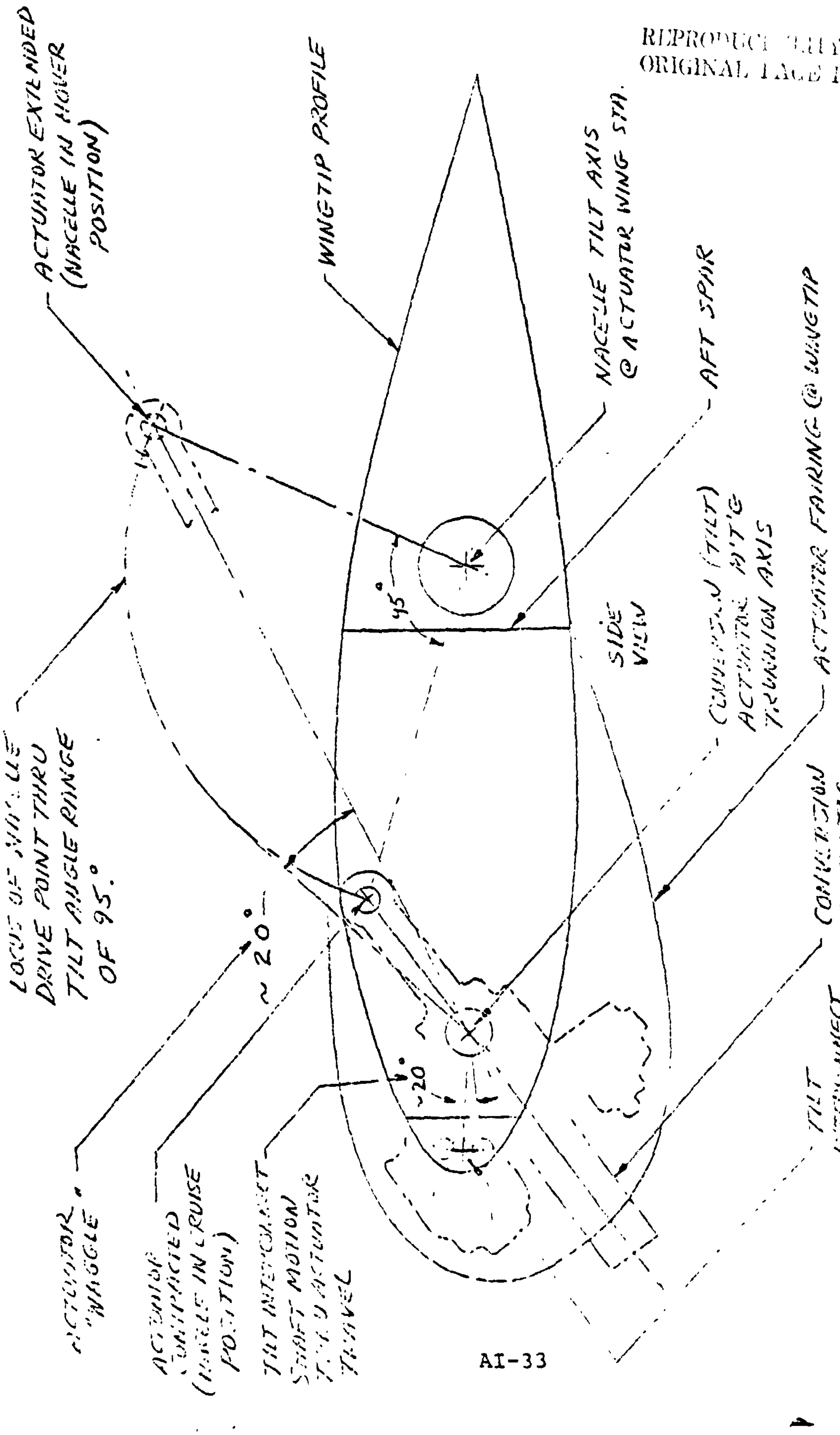


FIGURE A1.20. BASIC XV-15 CONVERSION (TILT) ACTUATOR - KINEMATICS (APPROX.)

TABLE A1.3

TRADE-OFF FACTORS - SPANWISE LOCATION OF NACELLE & CONVERSION ACTUATOR

OPTION NO.	1 (SEE FIG. 4)	2 (SEE FIG. 5)	3
TILT NACELLE SPANWISE LOCATION	AS XV-15 (C @ BL193)	MOVED OUTBOARD UP TO 6 IN.	MOVED OUTBOARD UP TO 6 IN.
CONVERSION (TILT) ACTUATOR SPANWISE LOCATION	AS XV-15	AS XV-15	MOVED OUTBOARD UP TO 6 IN.
ROTOR TIP-TO-FUSELAGE CLEARANCE (CRUISE FLIGHT)			
o Requirements			
- USAF (AFSC DH2-1; DN23C)	f (B,HP,M_{TIP}); Approx. 7.2 in. for B/V Rotor		
- USN -24K Vol. 1	12 in. Minimum		
- Civil FAR Part 25 Pg. 65	1 In. Min. Plus Distance Req'd to Prevent Harmful Vibr.		
o XV-15 A/C (25 Ft. Dia. Rotor)	12 in.	(Not Applic.)	(Not Applic.)
o B/V-Modified A/C (26 Ft. Dia. Rotor)	6 in.	7-12 in.	7-12 in.
SOUND PRESSURE LEVEL AT FUSELAGE WALL (CRUISE FLIGHT)			
o XV-15 A/C	(Later)	(Not Applic.)	(Not Applic.)
o B/V-Modified A/C	(Later)	(Later)	(Later)
DESIGN CONSIDERATIONS (B/V-Modified A/C)			
o Local Fuselage (Near Blade Tip) Beefup req'd	Uncertain	Less Probable	Less Probable
o XV-15 Wing Tip Changes Req'd	Little if any	Minor	Most Extensive; Build Out Structure
o Conversion (Tilt) System Changes Req'd (from XV-15)			
- Tilt Actuator Trunnion Support Components	None	Probably New-Increased Loads	None
- Wing Cross Shaft	None	None	Add Shaft Extensions
- Mechanical Backup Valve Actuation Line (to Actuator)	None	None	Add Extensions
- Electrical Lines to Actuator	None	None	Length Increases
- Actuator Output Arm Link to Tilt Nacelle	Little if any	New Longer Heavier Part	Little if any
o Tilt Actuator Out-of-Plane Torque on Tilt Nac. Struct.	Less	More	Less
o Tilt Nacelle Support Bearings Loads From Tilt Actuator	Less	More	Less
o Total Nacelle Structural Support Component Loads	Less	Greater	Greater
o A/C Roll Moment of Inertia	Least	Near Greatest	Greatest
o A/C Roll Control Power (Moment Arm)	Less	Greater	Greater
o A/C Empty Weight	Least	(Greater)	(Greater)
o Estimated Cost of Modifying Aircraft	Least	(Greater)	(Greater)

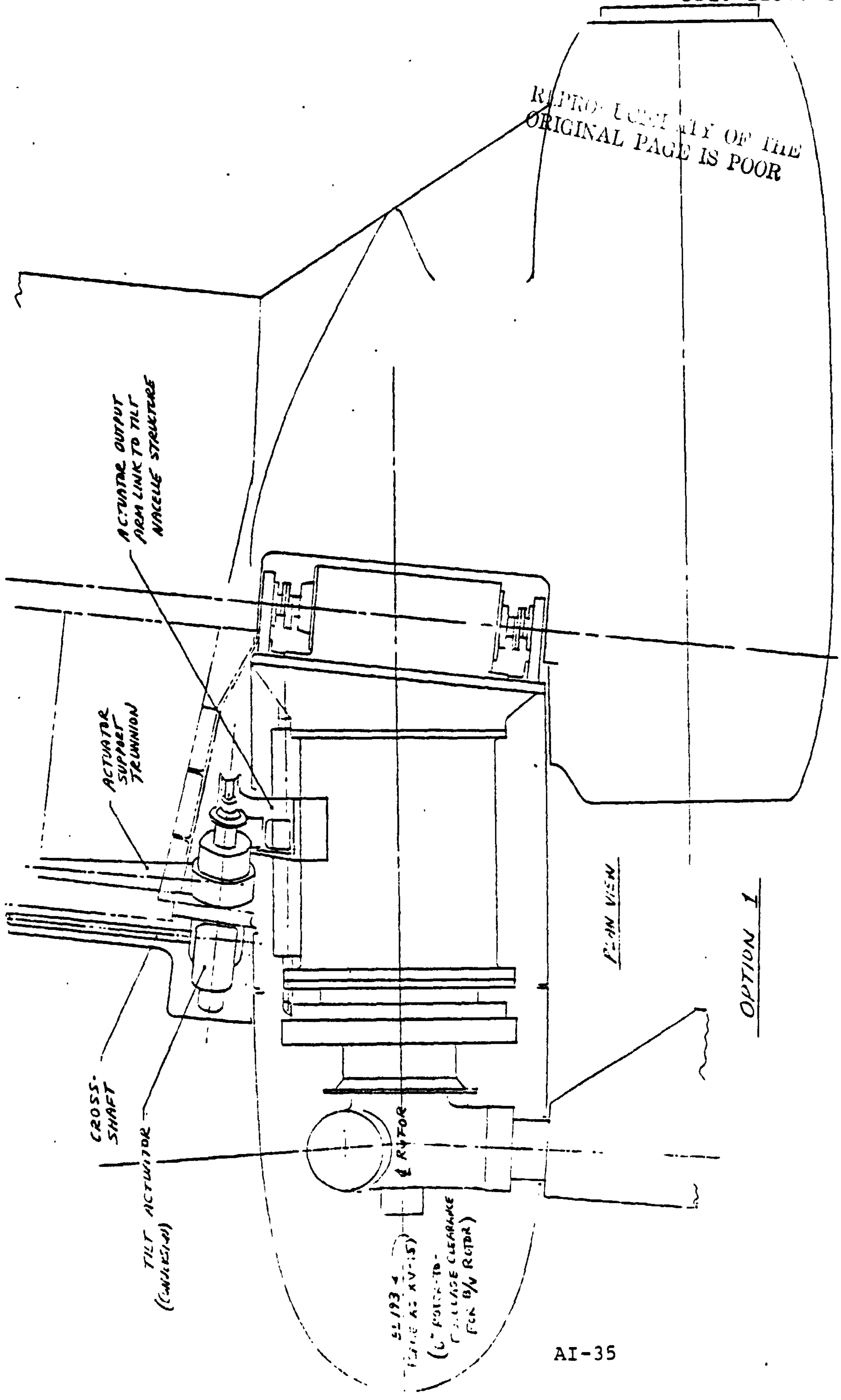


FIGURE A1.21. TILT NACELLE AND TILT ACTUATOR - SPANWISE LOCATIONS AS PER XV-15

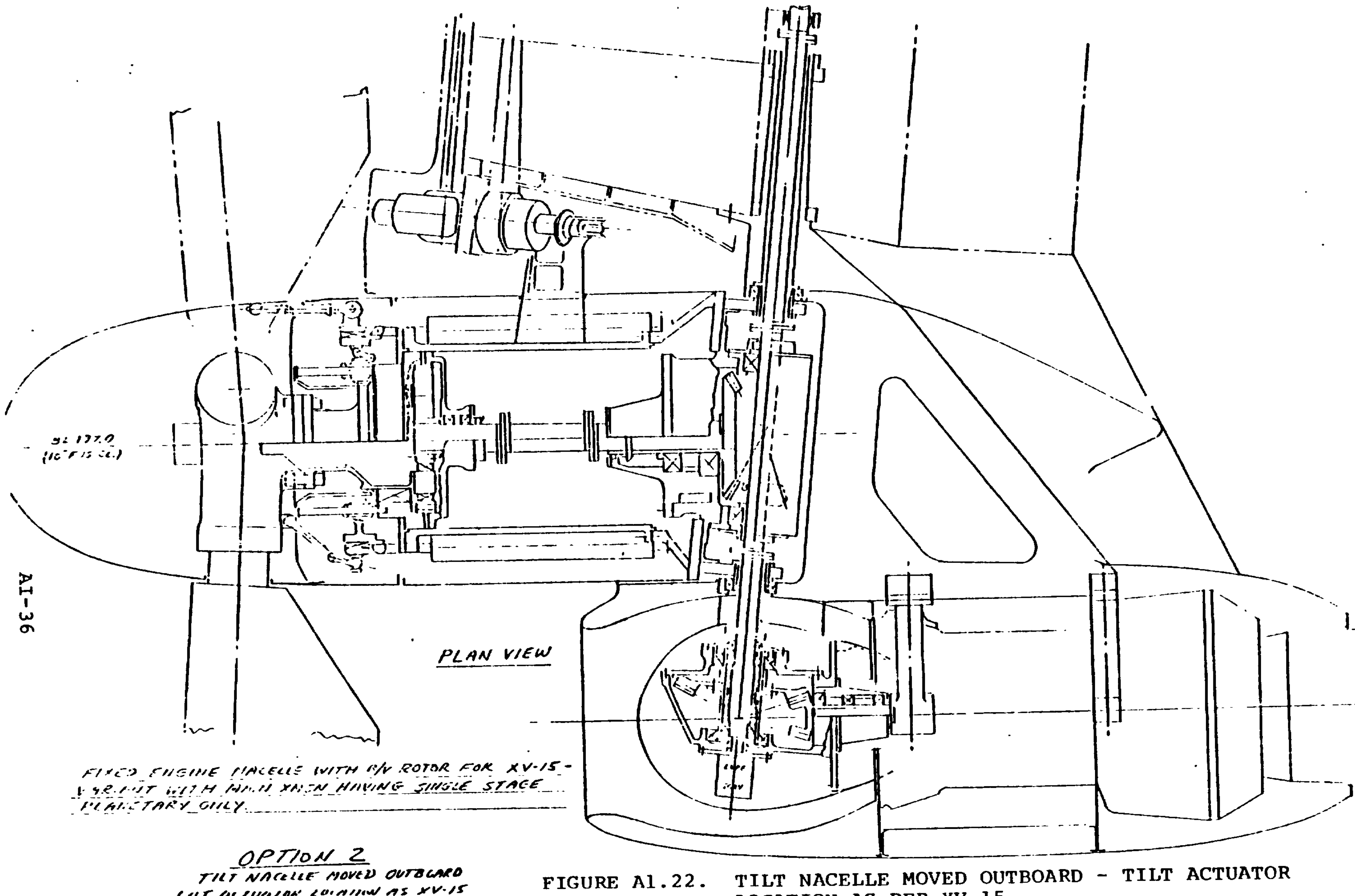


FIGURE A1.22. TILT NACELLE MOVED OUTBOARD - TILT ACTUATOR LOCATION AS PER XV-15

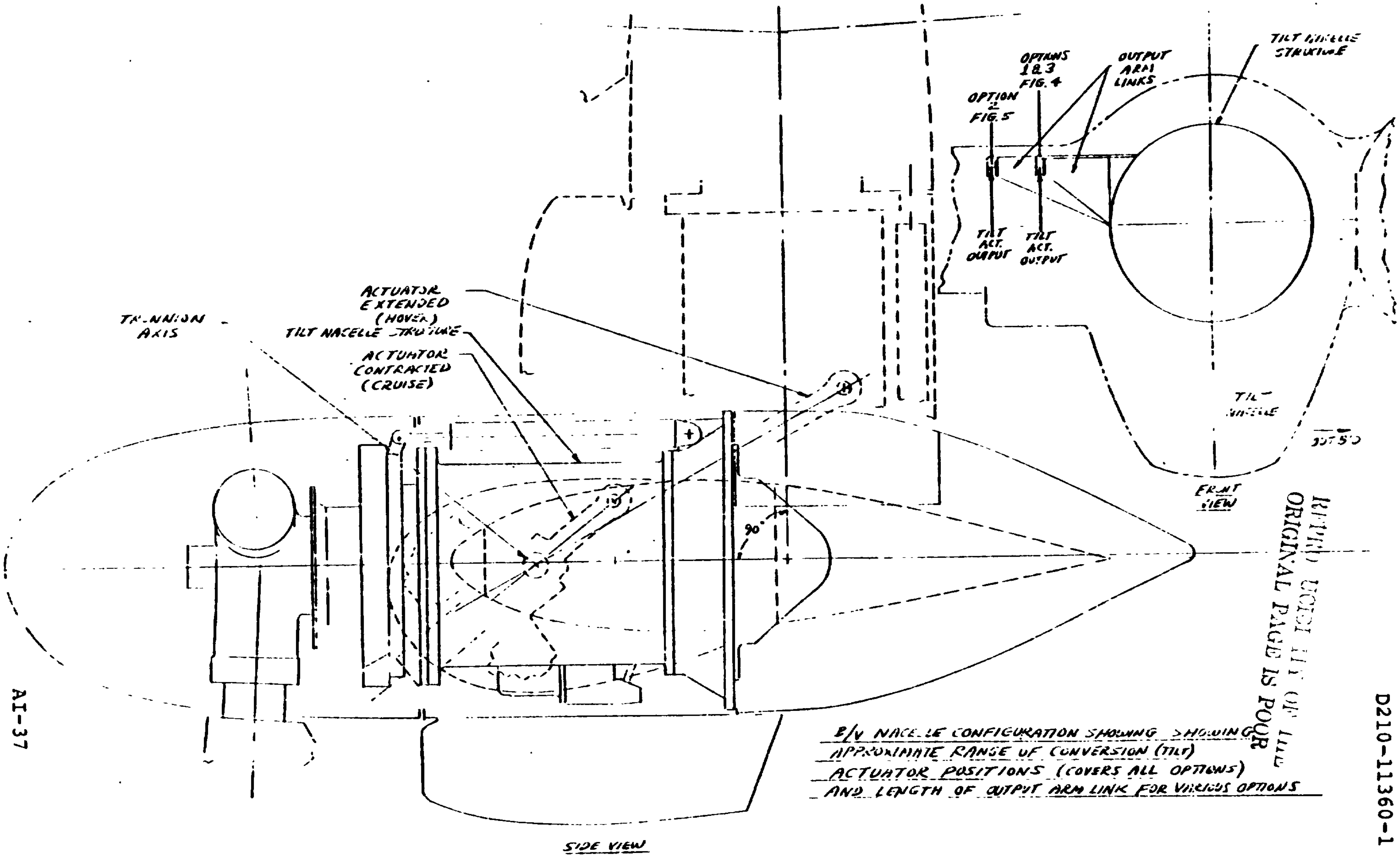


FIGURE A1.23. BOEING VERTOL NACELLE CONFIGURATION SHOWING APPROXIMATE RANGE OF CONVERSION (TILT) - ACTUATOR POSITIONS (COVERS ALL OPTIONS) AND LENGTH OF OUTPUT ARM LINK FOR VARIOUS OPTIONS

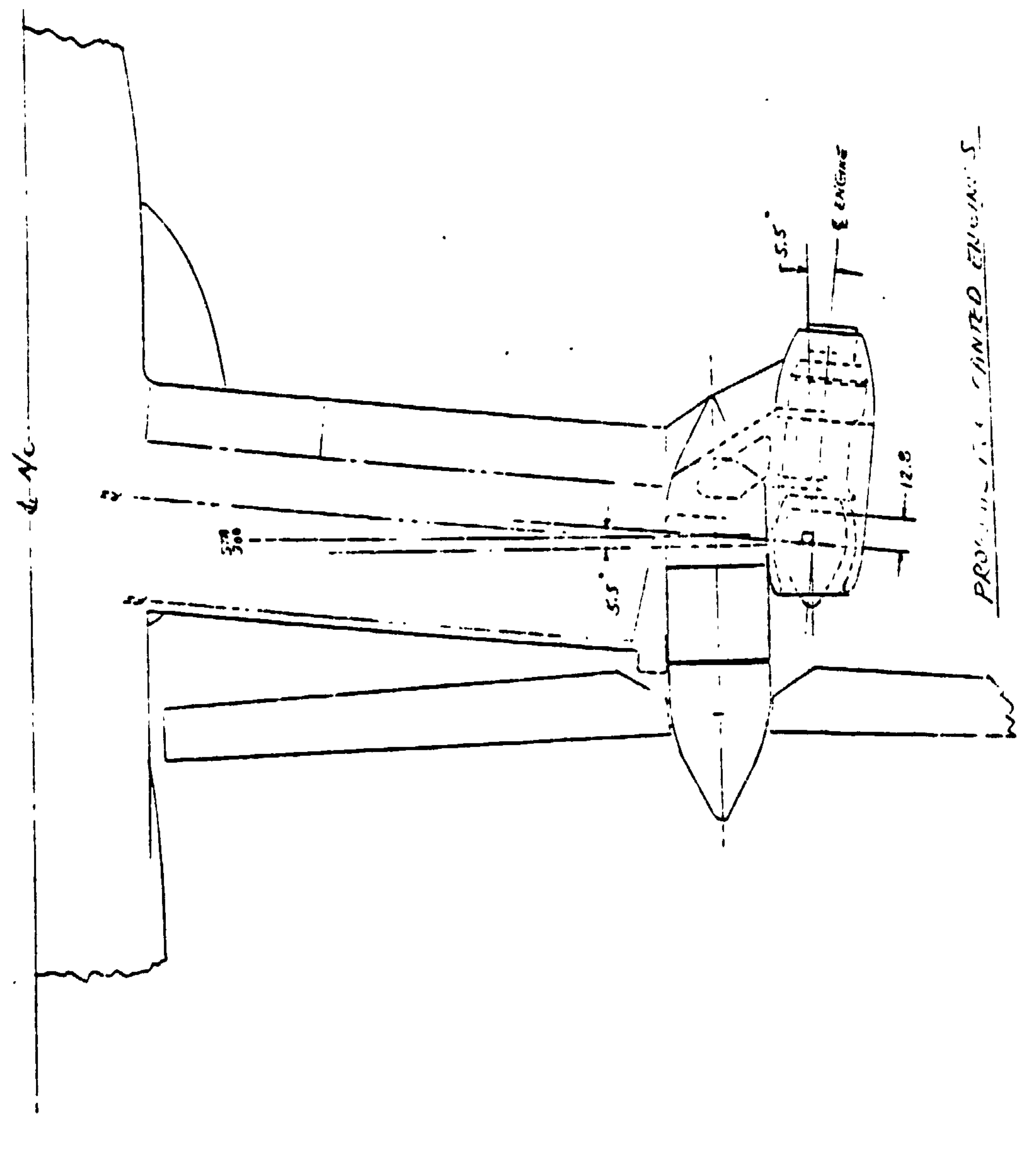


FIGURE A1.24. TOE IN FOR ENGINE INSTALLATION

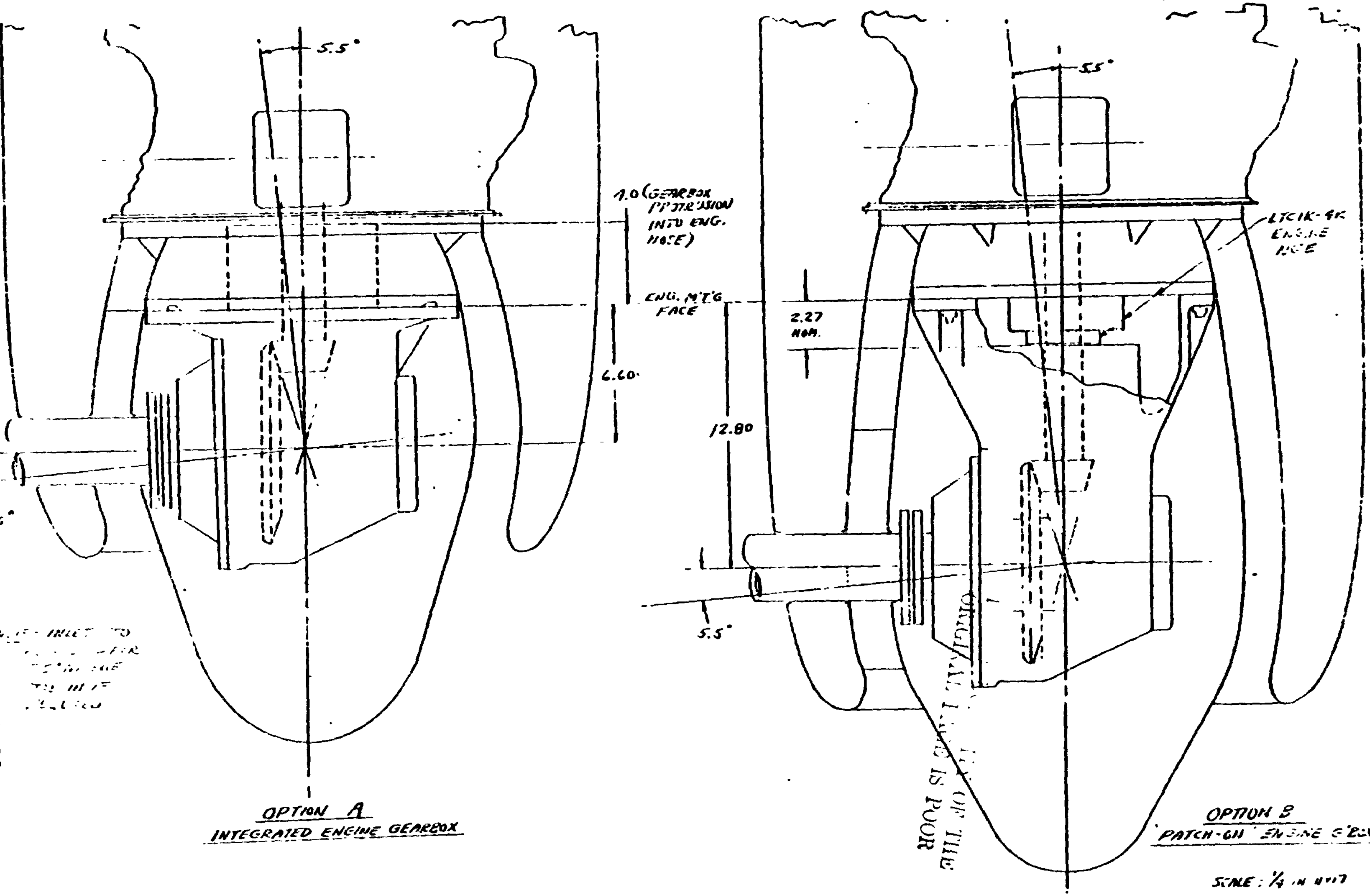


FIGURE A1.25. COMPARISON OF GEOMETRY FOR TWO-ENGINE NOSE/ ENGINE GEARBOX ARRANGEMENTS

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APPENDIX II - ROTOR AND HUB DESIGN

AII-1 - ROTOR HUB

The primary factor influencing the approach to the design of a rotor head for the HTR XV-15 was the experience gained since the Model 222 (Reference 3) was designed in 1971. In particular, over six years of effort was devoted to the design, manufacture and test of the YUH-61A UTTAS prototype (Reference 4) and to the design of the YUH-61A proposed production UTTAS. Over 2,000 flight hours were accumulated on four aircraft, the ground test vehicle ran an additional 1,500 hours and extensive bench testing was conducted.

A number of problems and their solutions were identified during this period. In addition, a comprehensive design improvement program, including weight, cost, maintainability and reliability considerations, was conducted while developing the production UH-61A rotor head configuration. Much of this information is applicable to a hingeless tilt rotor head.

The most significant departure from the Model 222 concept is in the area of blade retention. The blade is attached to the pitch shaft with two pins as opposed to the "coke bottle" retention of the Model 222. This greatly simplifies blade manufacture, eliminates all metal from the blade root end and permits the blade to be a separate component, removable independent of the rotor head. This latter characteristic enables inspection of individual components for fatigue damage and replacement without necessarily discarding the major elements of the blade.

AII-2 - DESIGN CONFIGURATION

General· The rotor hub assembly is shown in Figure A2.1. It is a three-bladed design, hingeless in the flap and chord axes, and provides .625 inch torque offset in the lead direction and 2.5 degrees precone from the center of rotation. The characteristics of the individual components are discussed in following sections. The material selected for each of the major components is listed in Table A2.1. To avoid repetition, as the various features of the design are discussed, they will be annotated to indicate the degree of Boeing Vertol experience with each.

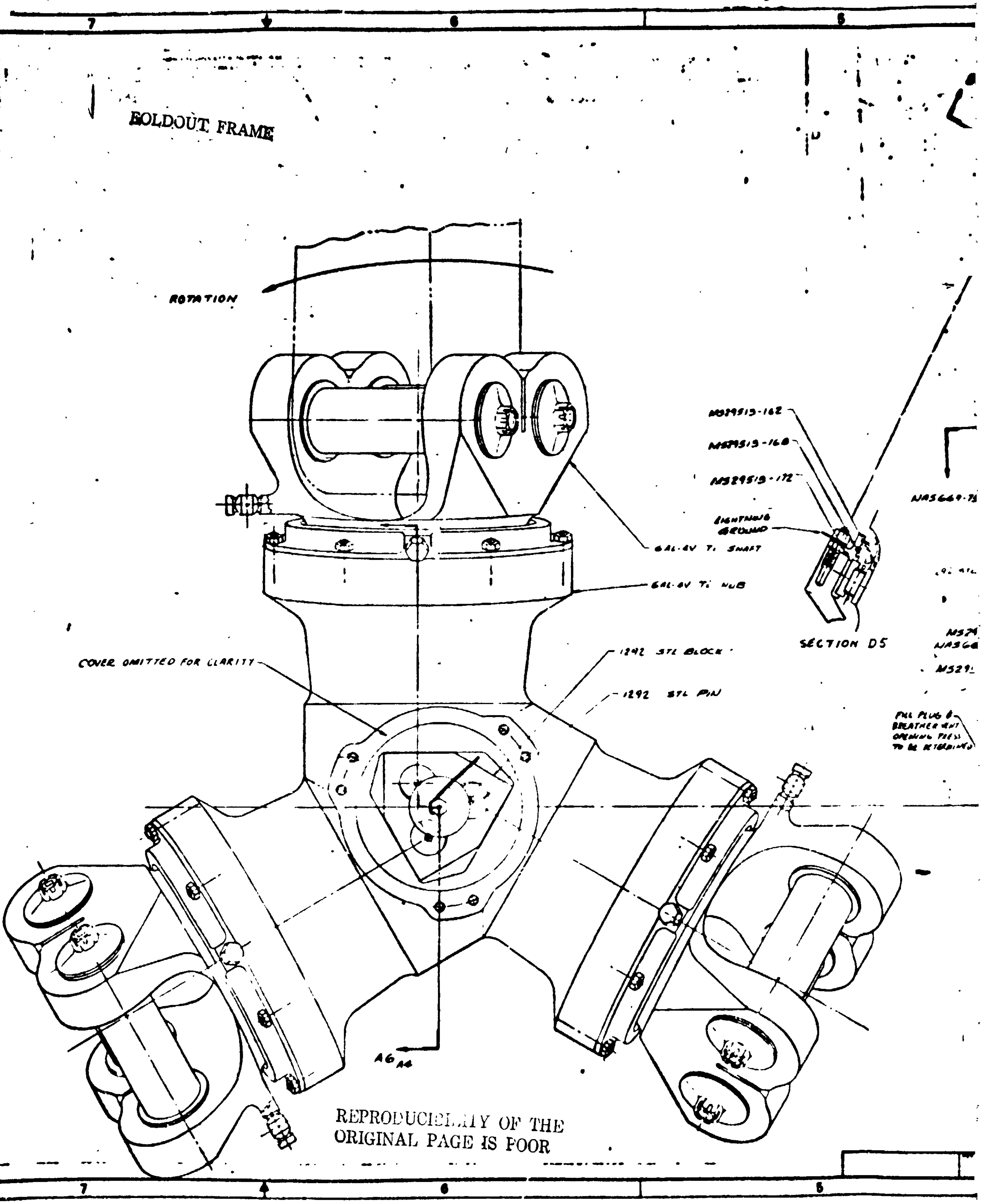
(1) Indicates used and tested on the Model 222.

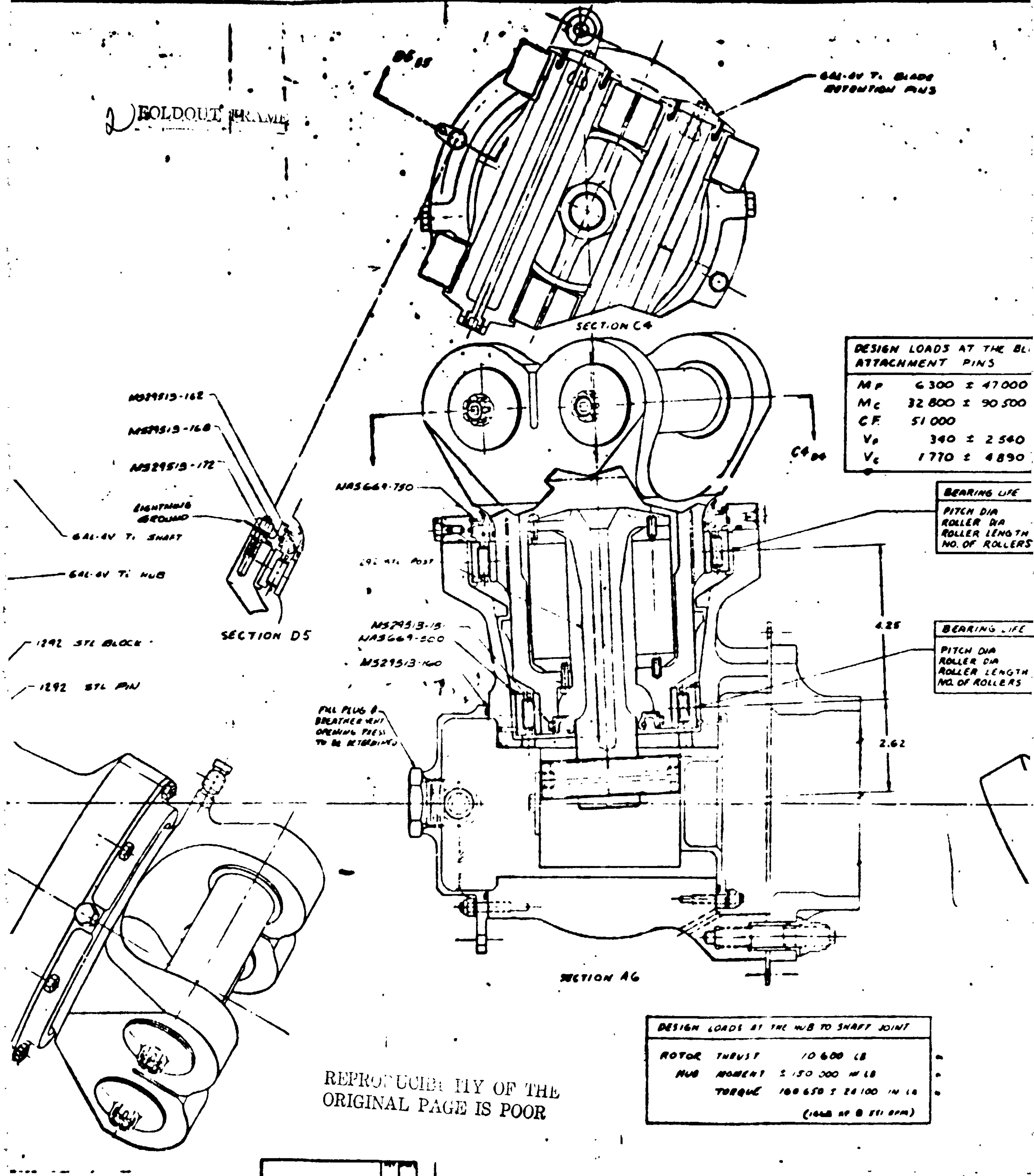
(2) Indicates feature used and tested on the YUH-61A.

(3) Indicates feature designed for the UH-61A.

Partially complete drawings of the major components are included as Figures A.2. and A2.3. However only the assembly drawing, Figure A2.1 fully depicts the final configuration.

Rotor Hub: The rotor hub is shown in Figure A2.2. The central hub is mounted to the main transmission rotor shaft with 12 studs and bushings. The studs react the rotor thrust and moment loads while the bushings transmit the torque(2).


ROTATION
MS29513-162
MS29513-168
MS29513-172
LIGHTNING GROUND
6AL-4V Ti SHAFT
6AL-4V Ti HUB
SECTION D5
COVER OMITTED FOR CLARITY
1292 STL BLOCK
1292 STL PIN
A6 A4

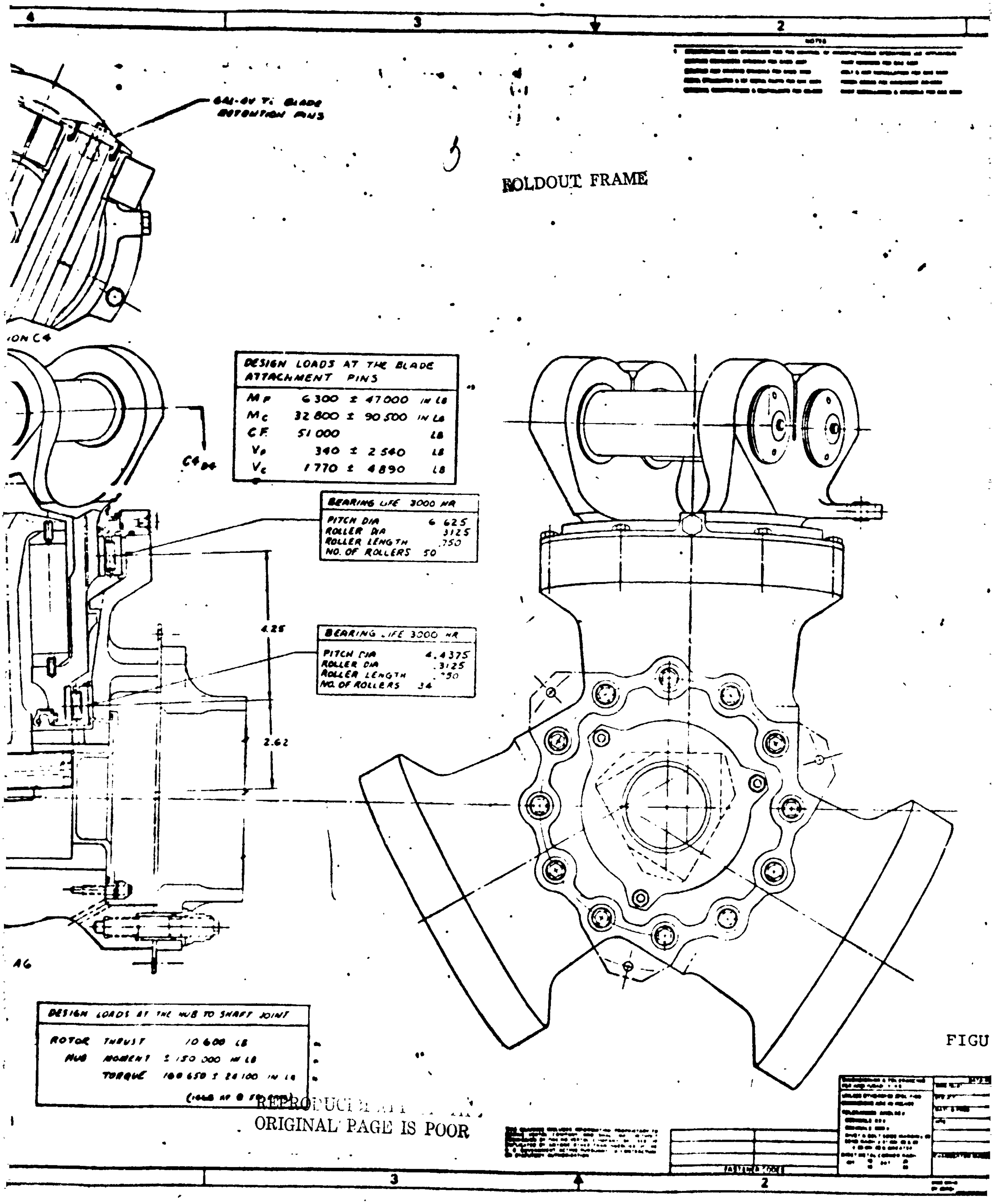


DESIGN LOADS AT THE BL ATTACHMENT PINS	
M_F	6 300 ± 47 000
M_C	32 800 ± 90 500
C.F.	51 000
V_F	340 ± 2 540
V_C	1 770 ± 4 890

BEARING LIFE
PITCH DIA
ROLLER DIA
ROLLER LENGTH
NO. OF ROLLERS

BEARING LIFE
PITCH DIA
ROLLER DIA
ROLLER LENGTH
NO. OF ROLLERS

DESIGN LOADS AT THE HUB TO SHAFT JOINT		
ROTOR	THRUST	10 600 LB
HUB	MOMENT	± 150 000 IN LB
	TORQUE	100 650 ± 24 100 IN LB
		(LOAD AT [illegible] RPM)



DESIGN LOADS AT THE BLADE ATTACHMENT PINS		
M_F	6 300 ± 47 000	IN LB
M_C	32 800 ± 90 500	IN LB
C.F.	51 000	LB
V_F	340 ± 2 540	LB
V_C	1 770 ± 4 890	LB

BEARING LIFE 3000 HR	
PITCH DIA	6.625
ROLLER DIA	.3125
ROLLER LENGTH	.750
NO. OF ROLLERS	50

BEARING LIFE 3000 HR	
PITCH DIA	4.4375
ROLLER DIA	.3125
ROLLER LENGTH	.750
NO. OF ROLLERS	34

DESIGN LOADS AT THE HUB TO SHAFT JOINT			
ROTOR	THRUST	10 600	LB
HUB	MOMENT	± 150 000	IN LB
	TORQUE	100 650 ± 26 100	IN LB

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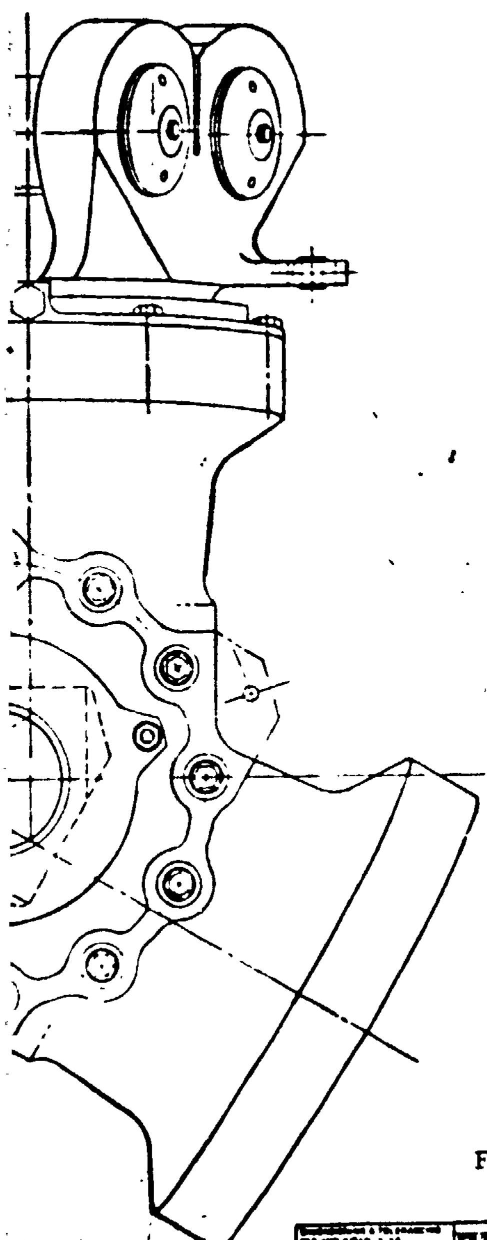

FIGURE A2.1. HUB ASSEMBLY

ROTOR HEAD ASSY-
NTR XV-15

J 77272 SK27252

SK27252

COMPONENT	MATERIAL	SPECIFICATION	TENSILE STRENGTH-KS
ROTOR HUB & LINERS	TITANIUM	MIL-T-9047	130
PITCH SHAFT	TITANIUM	MIL-T-9047	130
CENTER BLOCK	CRES	AMS 5617	205
RETNETION POST	CRES	AMS 5617	205
RETENTION PIN	CRES	AMS 5617	205
BLADE ATTACHMENT PIN	TITANIUM	MIL-T-9047	130
PIN BOLT	TITANIUM	MIL-T-9047	130
PIN CAP	TITANIUM	MIL-T-9047	130
OUTBOARD BEARING	STEEL	AMS 6491	---
INBOARD BEARING	STEEL	AMS 6491	---
MOUNT BUSHING	STEEL	AMS 6414	150
MOUNT STUD	STEEL	AMS 6414	150
MOUNT SPACER	CRES	MIL-S-25043	177
BEARING RETAINER	ALUMINUM	QQ-A-250/12	68
LOWER POSITIONER	ALUMINUM	QQ-A-225/9	68
RESERVOIR	ALUMINUM	QQ-A-250/12	65

TABLE A2.1. ROTOR HEAD COMPONENT MATERIALS SELECTION

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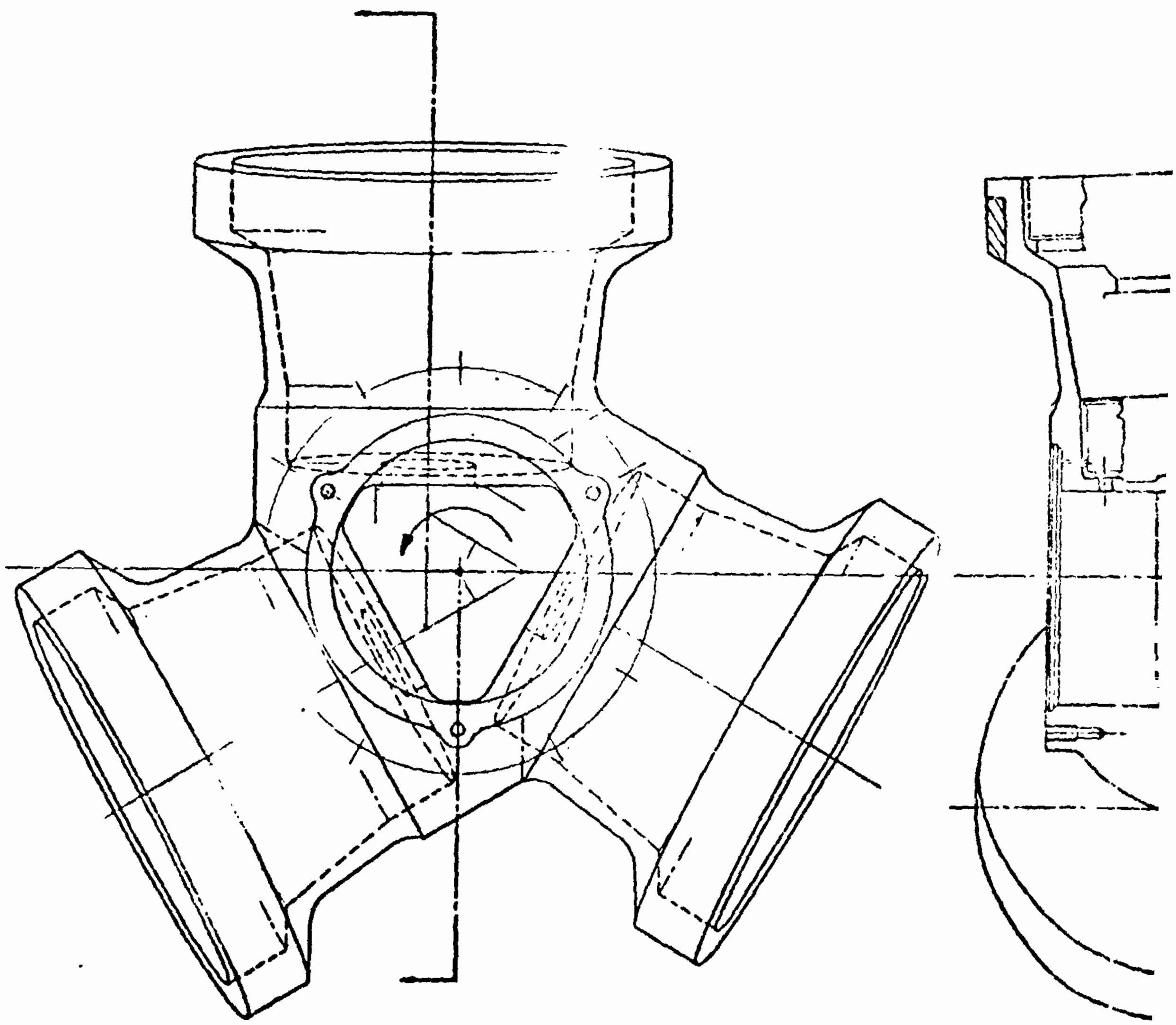

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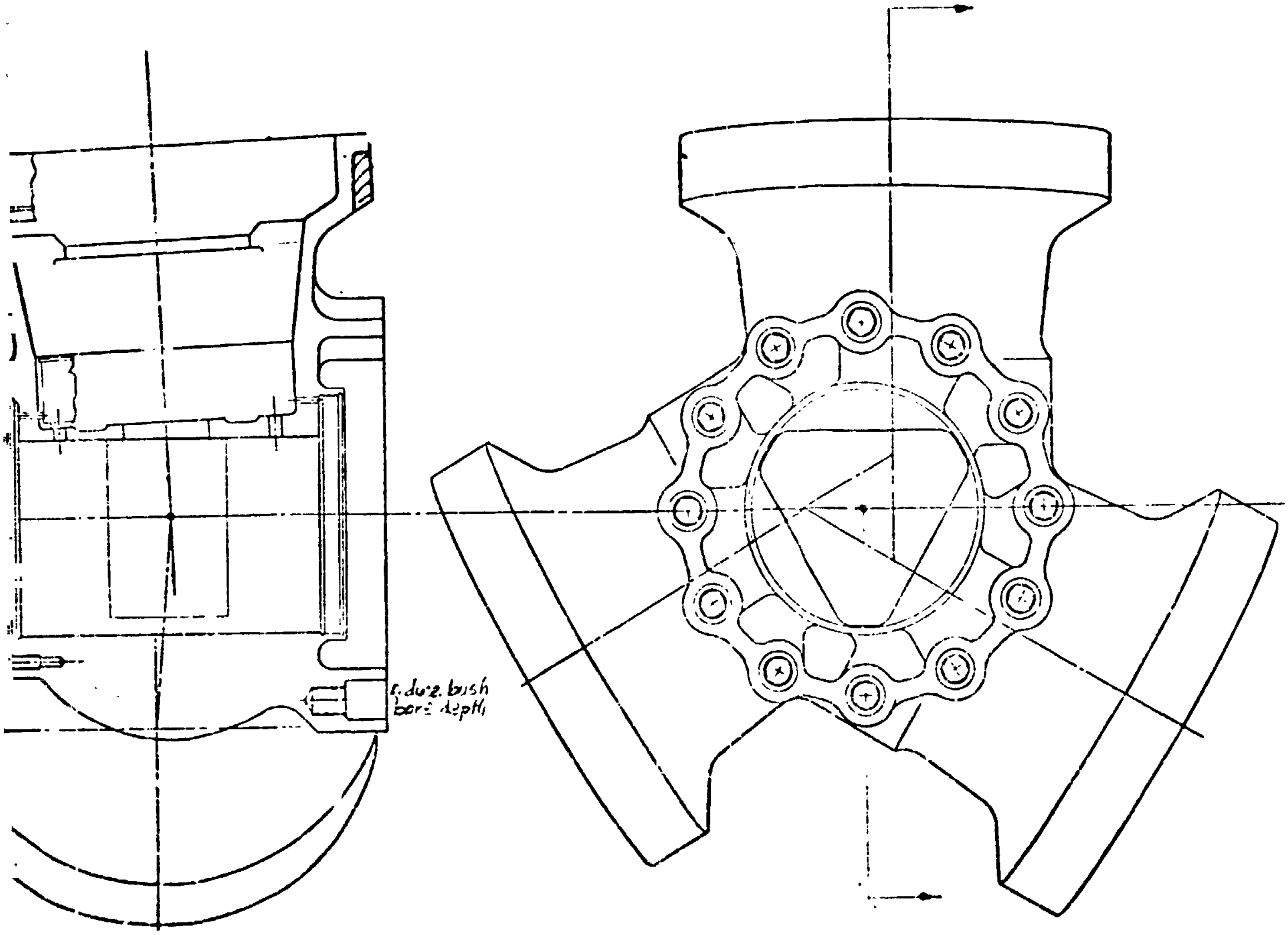

FIGURE A2.2. ROTOR HUB

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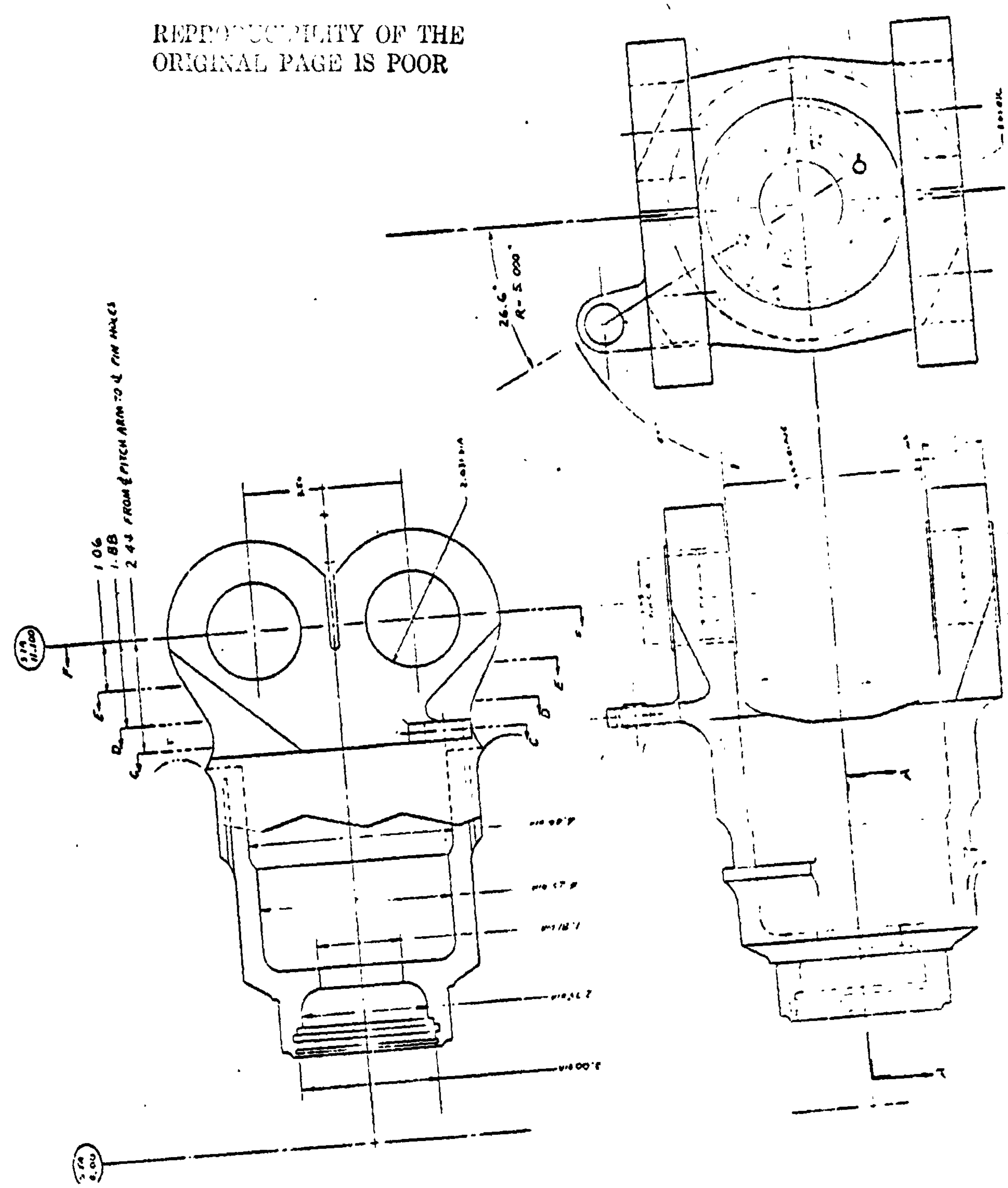


FIGURE A2.3. PITCH SHAFT

The connection is redundant in that it is designed to function with several studs and/or bushings failed. An aluminum-bronze-ekonol coated spacer[2] is installed between the hub and shaft to prevent fretting. The spacer also serves as an attachment for the spinner support.
The three pitch housings are integral with the hub [2]. Bearing liners are installed in the pitch housings to prevent fretting and also to permit bearing installation and removal without damage to the hub[2].

Pitch Shaft: The pitch shaft is shown in Figure A2.3. The inboard barrel section is mounted within the hub pitch housing on a pair of roller bearings[2]. There is no pitch horn as such; the pitch link connects directly to a lug on the pitch shaft. At its outboard end, the pitch shaft transitions to a clevis to provide the blade attachment[2].

Bushings are installed in the lugs to prevent fretting and to permit pin installation and removal without damage to the pitch shaft[2].

Blade Attachment: The rotor blade is mounted within the pitch shaft clevis and attached with two hollow pins[3]. The outside diameter of the pin is stepped to facilitate installation and removal[2]. Clamp-up is provided by a bolt through the center of the pin which threads into a cap nut[3]. An additional nut on the bolt traps it within the pin and permits the bolt to be used as a pin puller[3]. To prevent fretting the section of the pin within the pitch is tungsten-carbide coated[2] while the section within the blade is coated with teflon-dacron fabric[3].

With the clevis and two pin arrangement, there are four load paths between the blade and pitch shaft. The connection is redundant in that it is designed for 30 hours life with any one load path missing due to a pitch shaft or pin failure[3]. The clevis is slotted between the holes to prevent a crack from propagating from one lug to the other[3].

Pitch Bearings & Lubrication System: Pitch change motion between the pitch shaft and hub is accommodated by a pair of oil-lubricated roller bearings[2]. The bearings are designed with two roller per cage pocket to maximize capacity[2], and both bearings have a B-10 life of 3,000 hours.

The oil reservoir is located on the top of the hub[3]. It is fitted with a filling port and a sight glass for ascertaining proper oil level. Two ports are provided in the outboard bearing retainer[2]. The upper one permits air to escape during filling while the lower one is used to drain the oil.

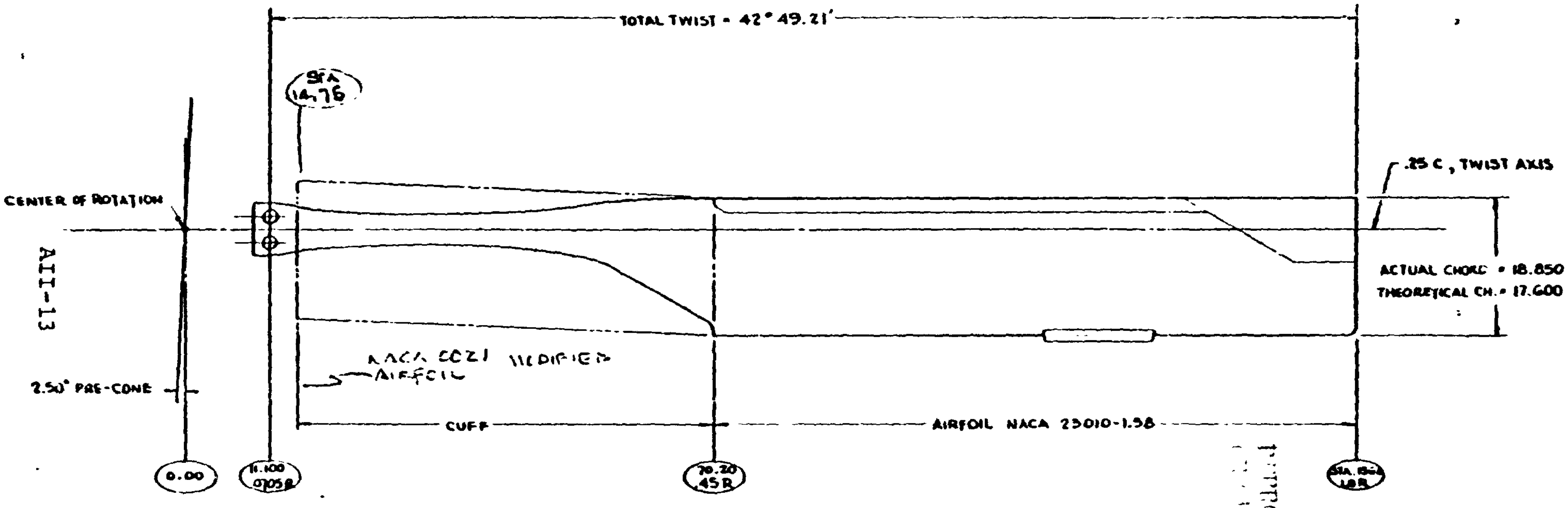
The lubrication system serves as a failsafe mechanism for the hub and pitch shaft[2]. Both are designed for 30 hours life after a crack has propagated through the wall and permitted the oil to escape[2]. Thus, a loss of oil, will indicate the possibility of a crack. In addition, a series of 8 holes are drilled in the wall of the pitch shaft under the outboard bearing to extend the failsafety to this area.

<u>CF Retention System</u>: The rotor blade centrifugal force is transmitted through the attachment pins to the pitch shaft and then to laminated elastomeric bearing(1). The bearing is retained by a post(1) which is pinned to a block in the center of the hub. The post and block are shown in Figure A2.1. Redundancy for the CF retention components is provided by a pair of flanges which extend over opposite 90° arcs on the hub inside diameter and pitch shaft outside diameter(3). The pitch shaft is installed in "bayonet" fashion, i.e., the shaft is inserted in the hub and rotated 90° to engage the flanges. Normally the flanges are separated, but come into contact to prevent the loss of the rotor blade should a retention component fail. Since the flanges are located within the oil cavity, lubrication is provided to the surfaces in contact to accommodate pitch change motions.

AII-3 - <u>BLADES & CUFF</u>

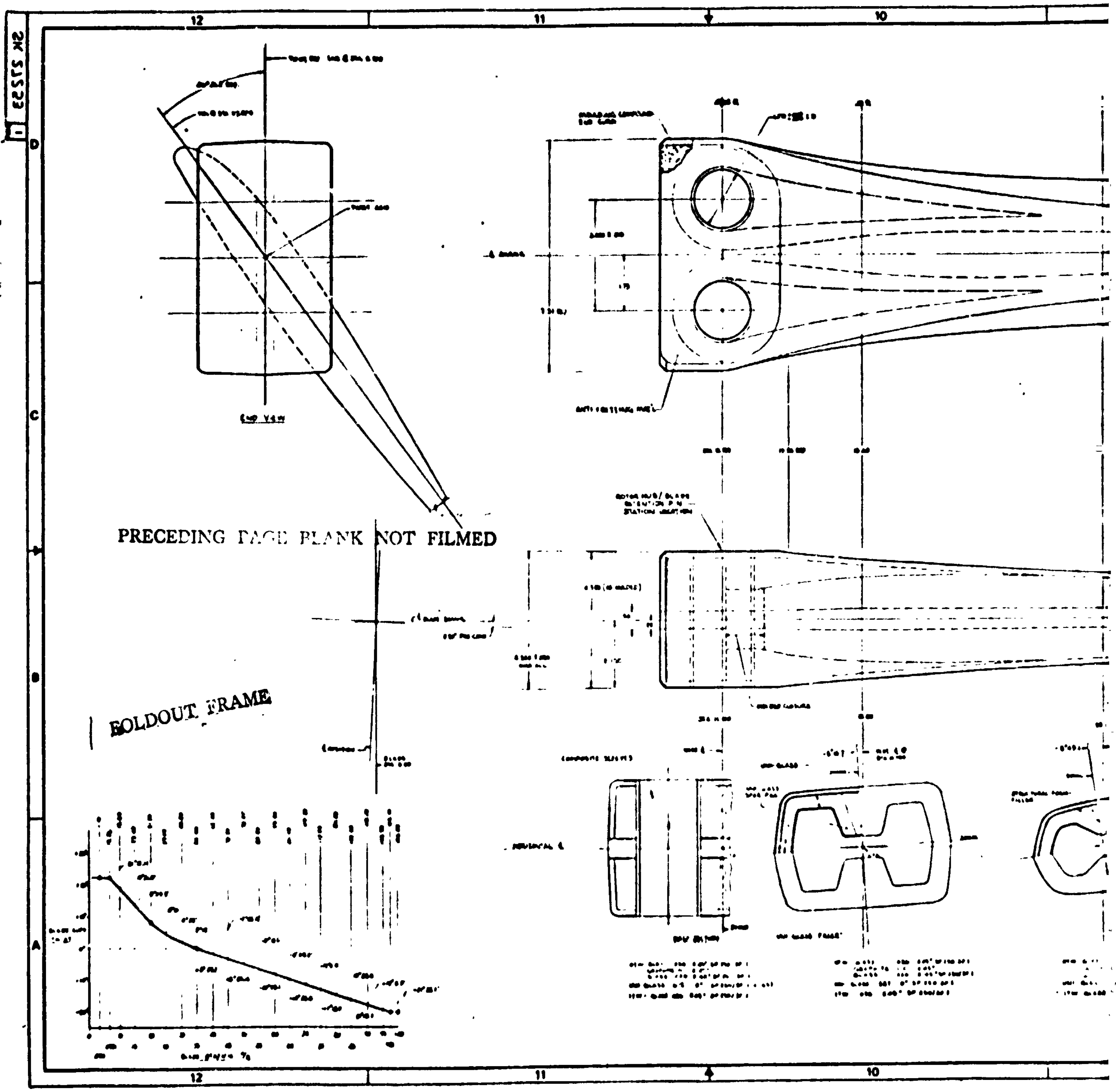
Blade Description: SK-27253 (Figures A2.4 to A2.7), shows the HTR XV-15 rotor blade, configured to meet the objectives and criteria of this R&D effort. The two pin attached, composite blade, embodies much of today's state-of-the-art, proven design features and fabrication processes.

The two-pin blade to hub attachment is located at Station 11.100 (.071 R). The blade radius remains as it was on the Model 222, Station 156.00. All boron cross-plied materials have been replaced by graphite cross-ply. This includes the



HTR-XV15 ROTOR BLADE
(UNTWISTED PLANFORM)

FIGURE A2.4. TILT ROTOR BLADE - MODEL HTR XV-15



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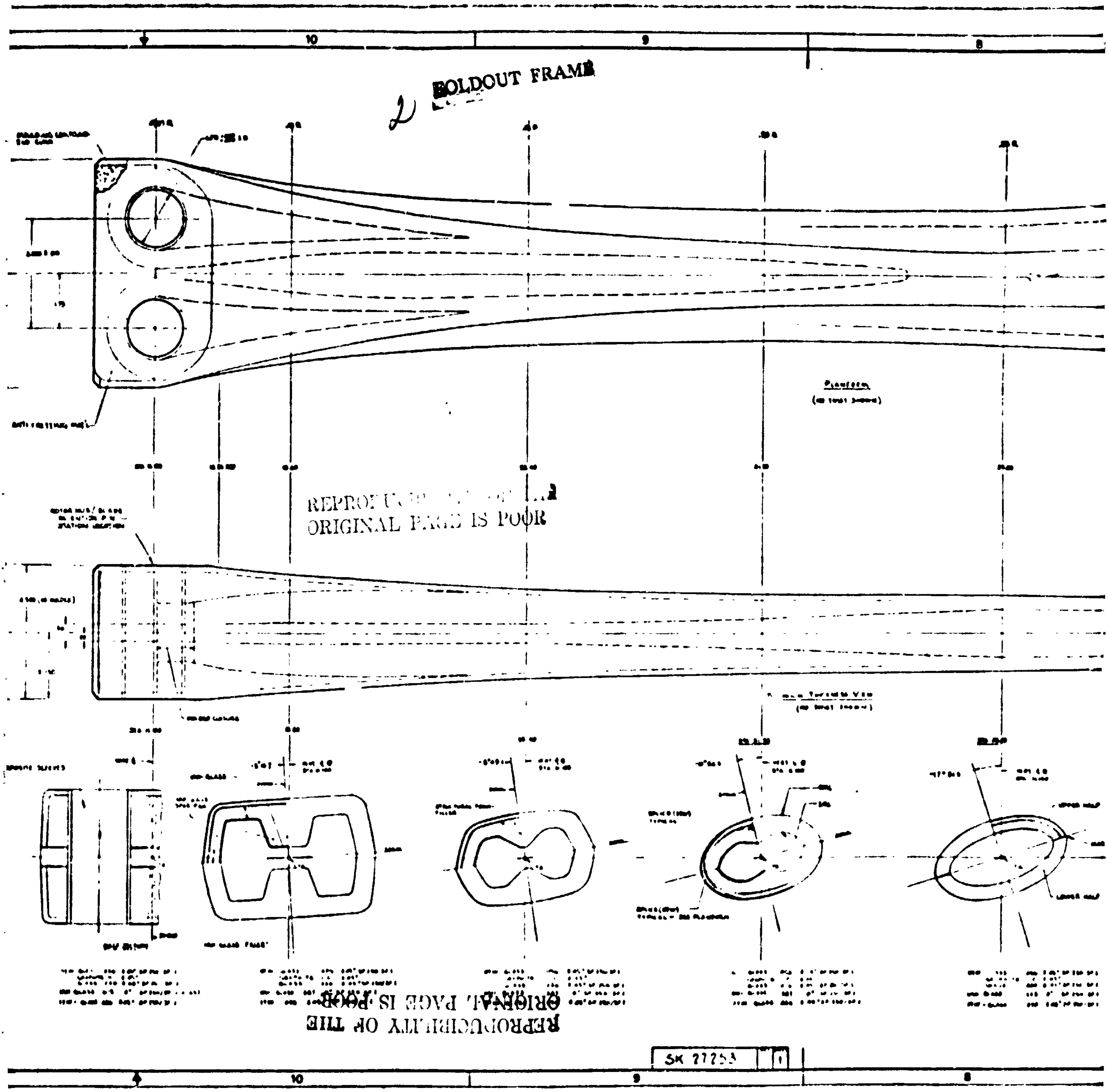

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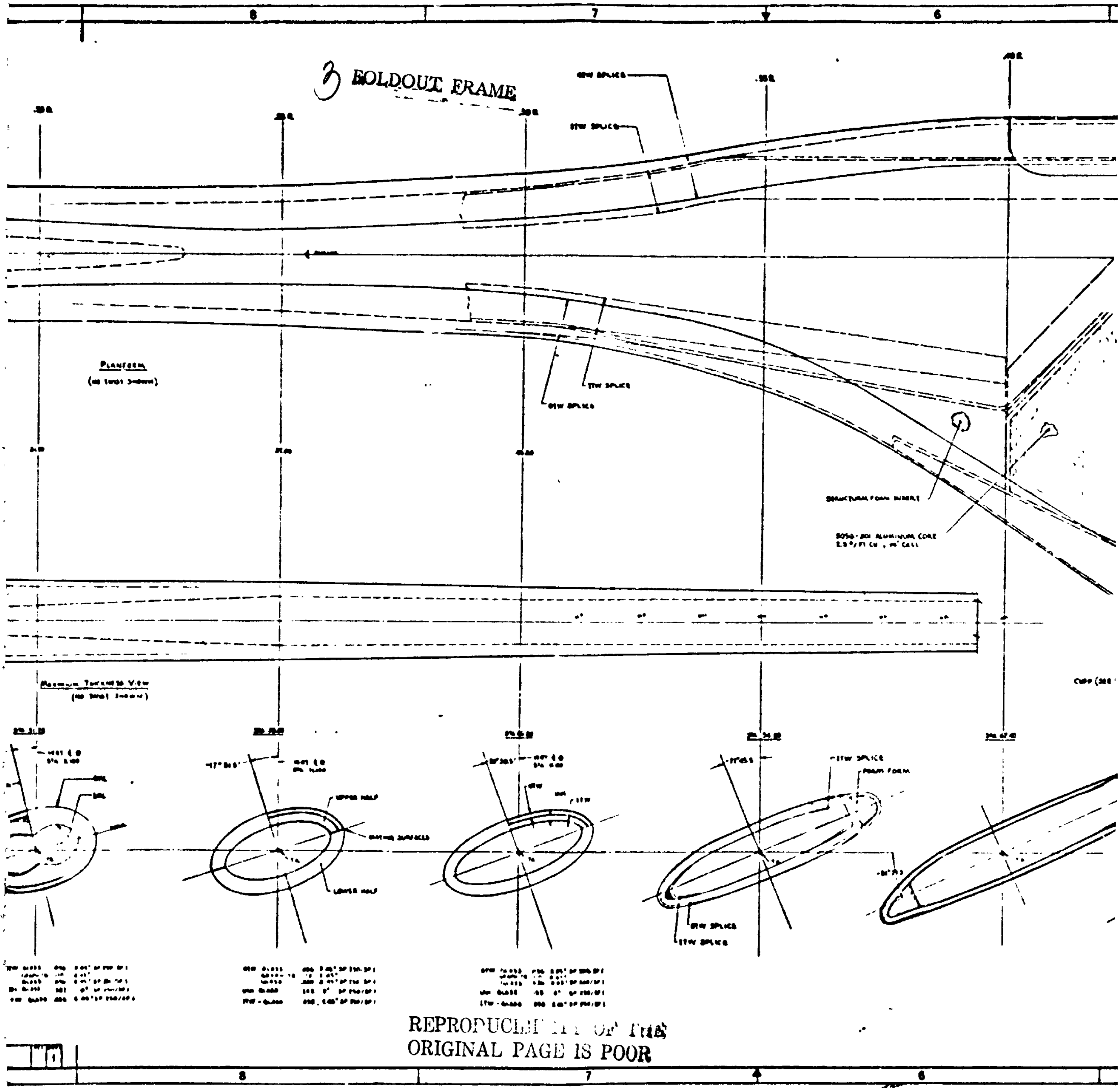

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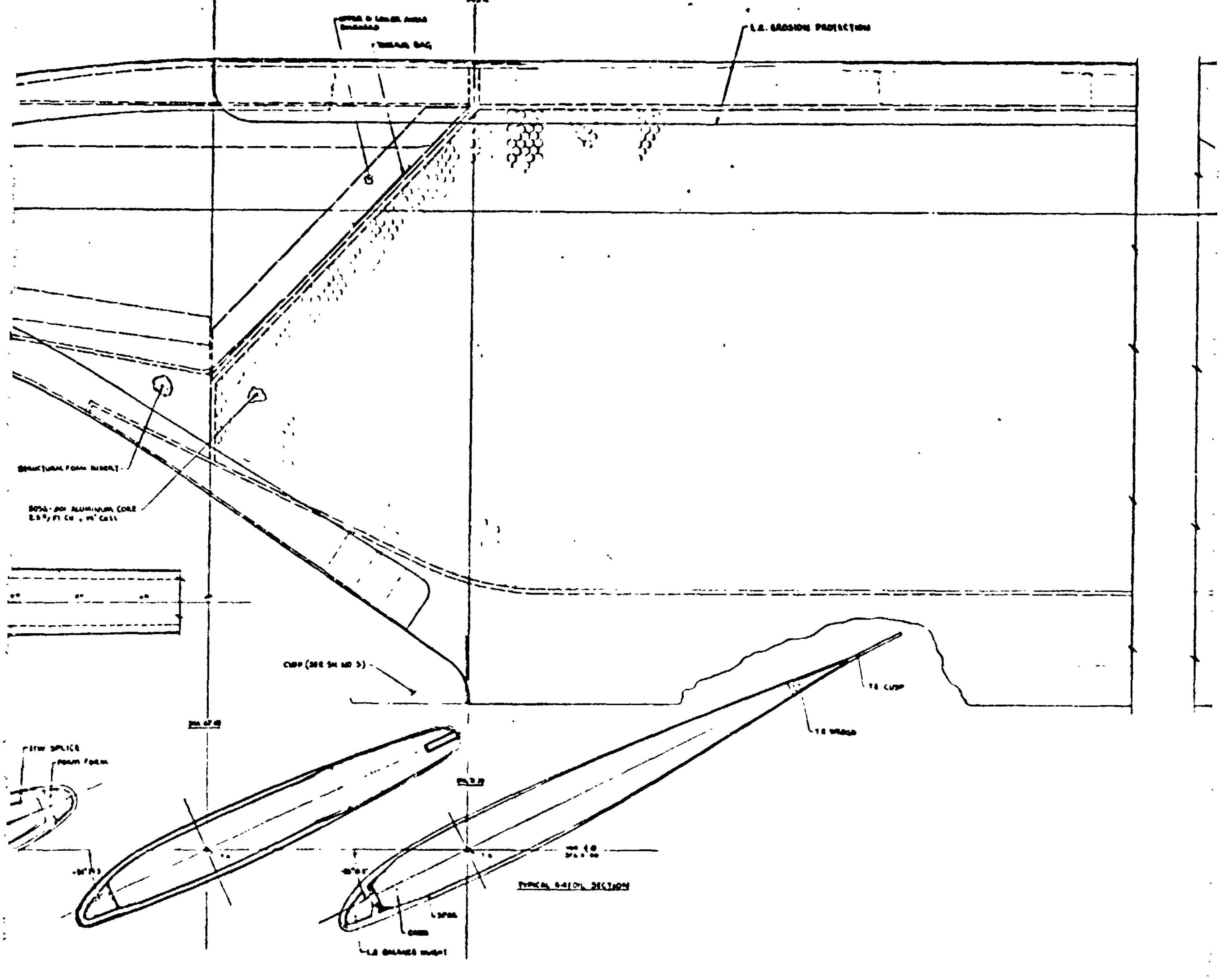

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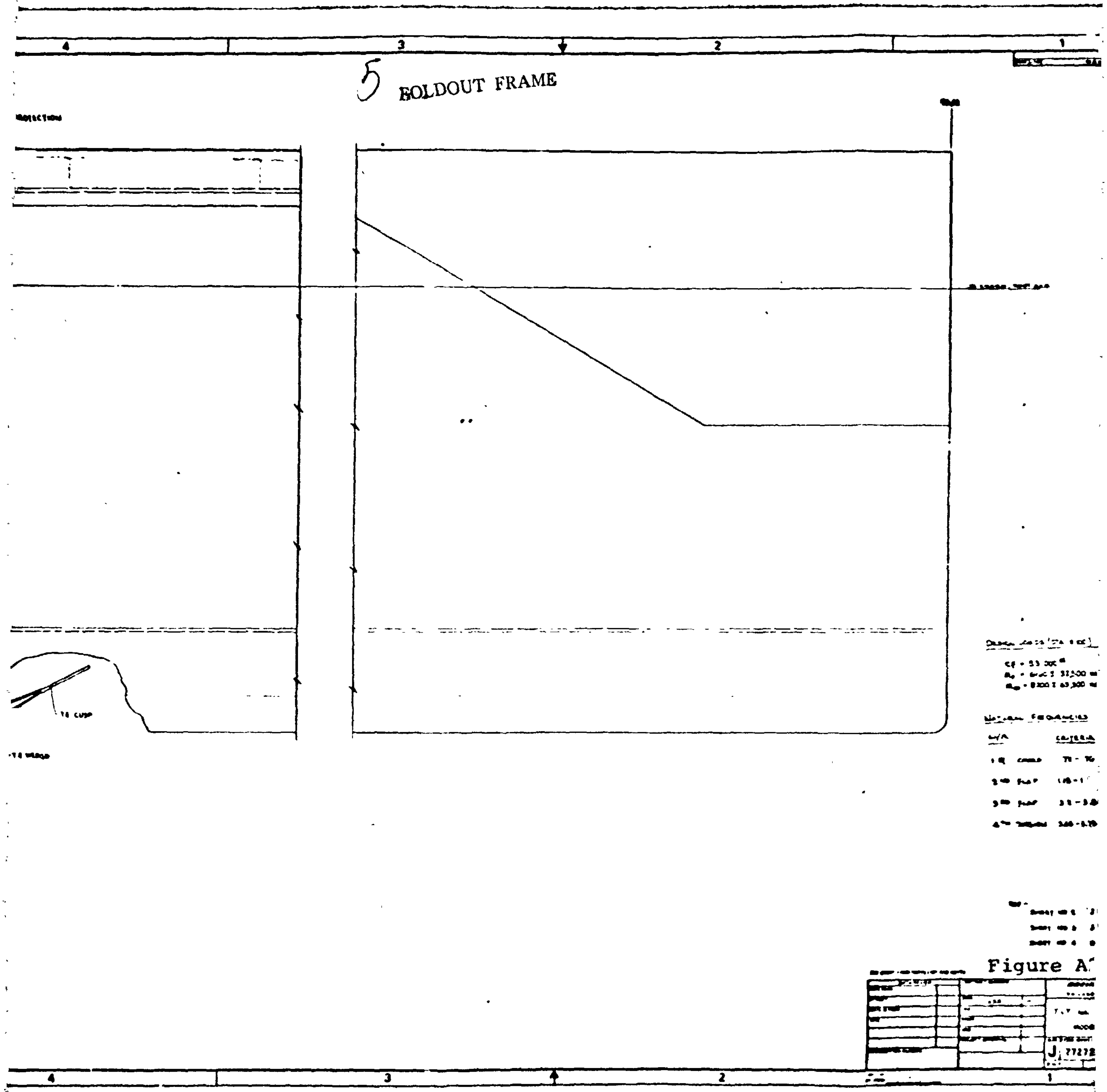

Figure A

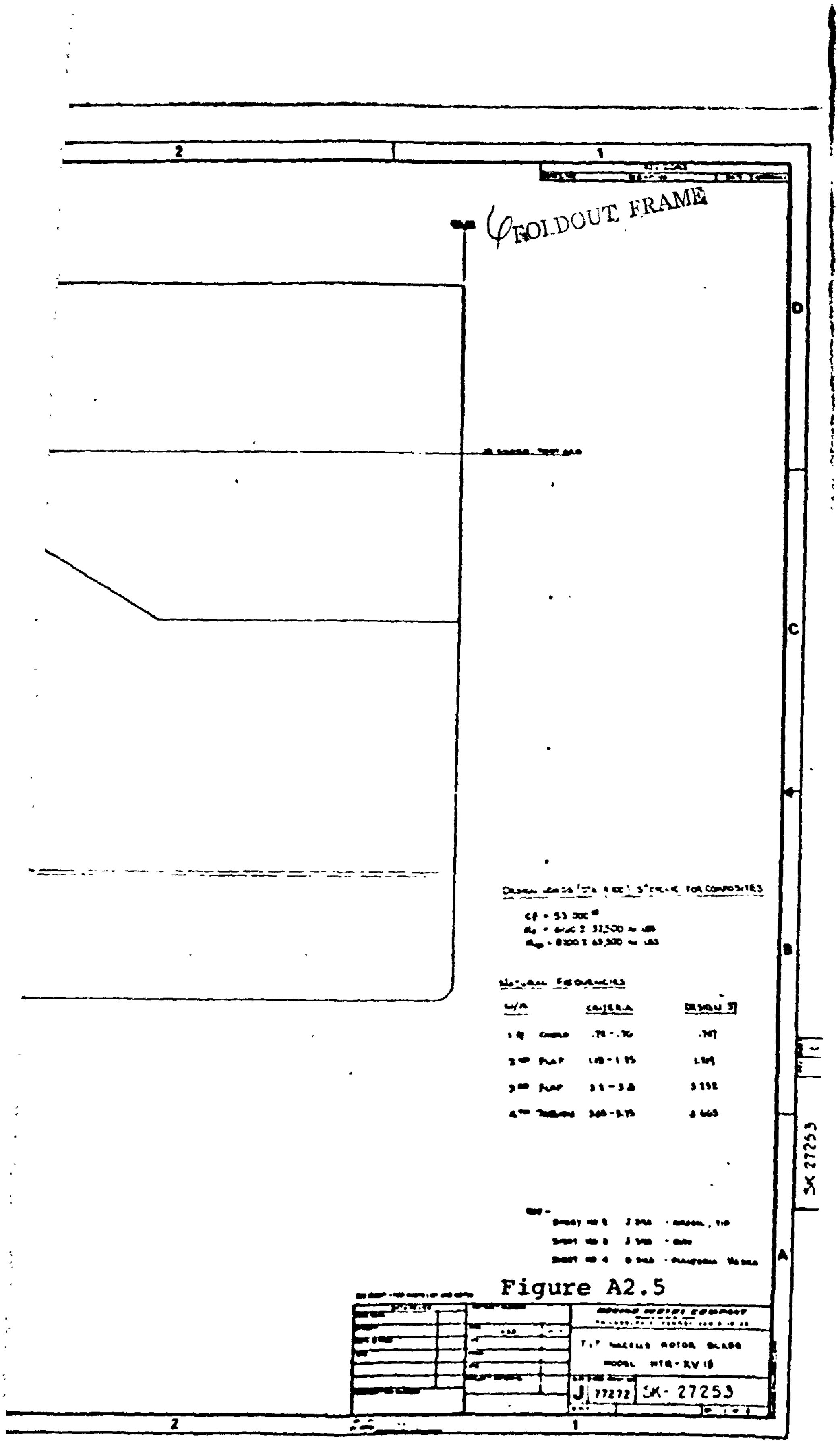

Figure A2.5

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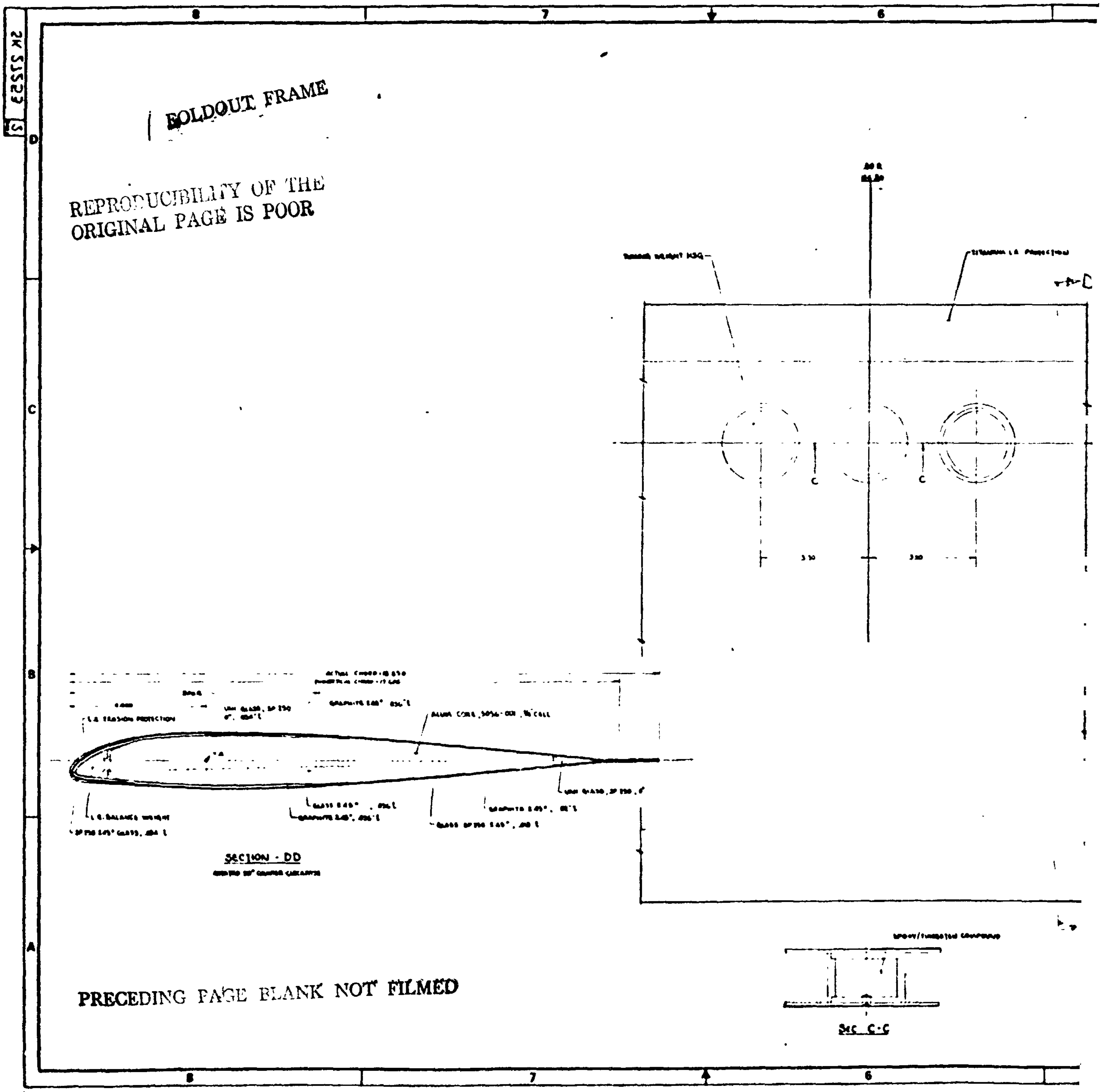

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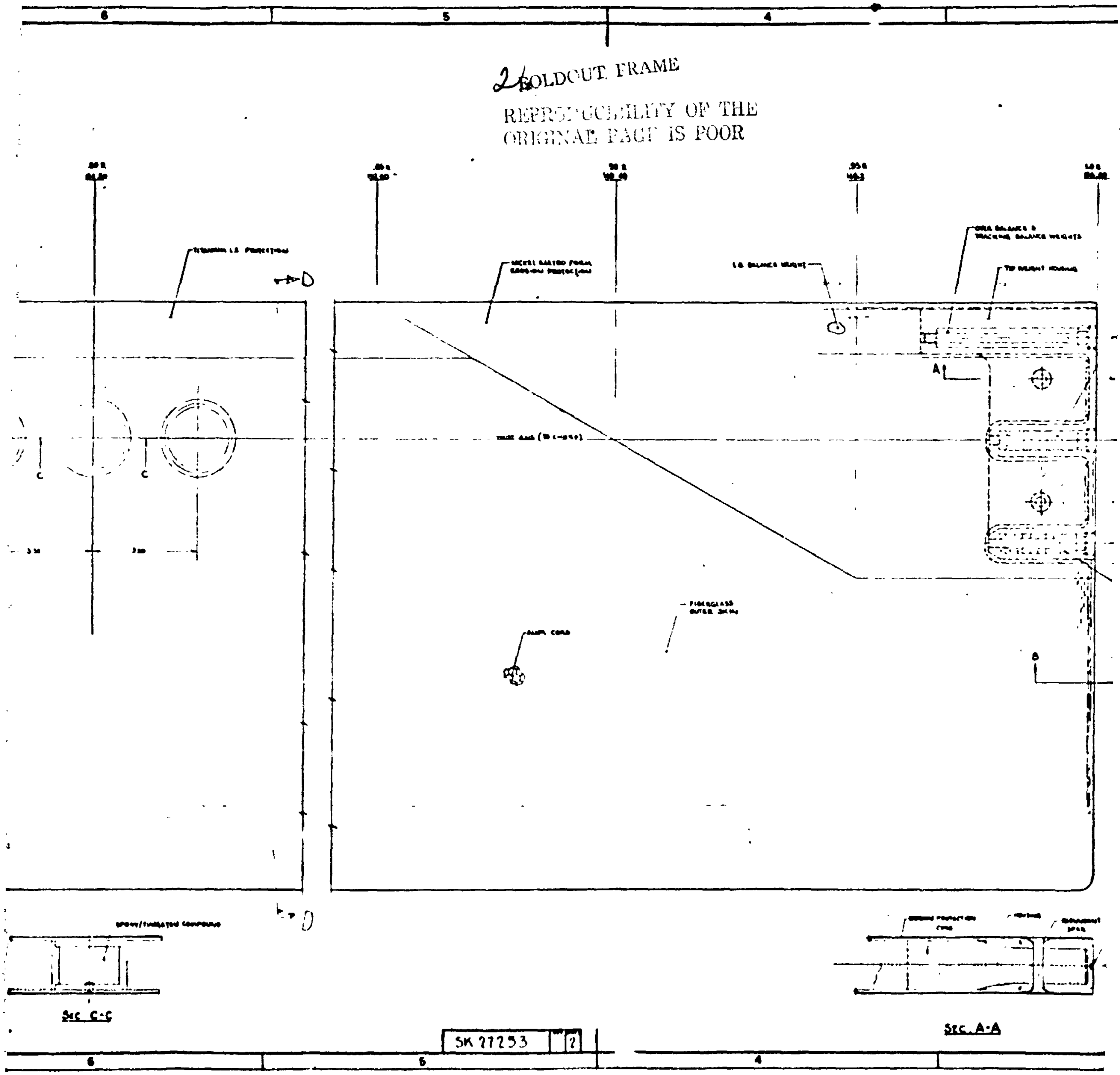

SK 27253 2

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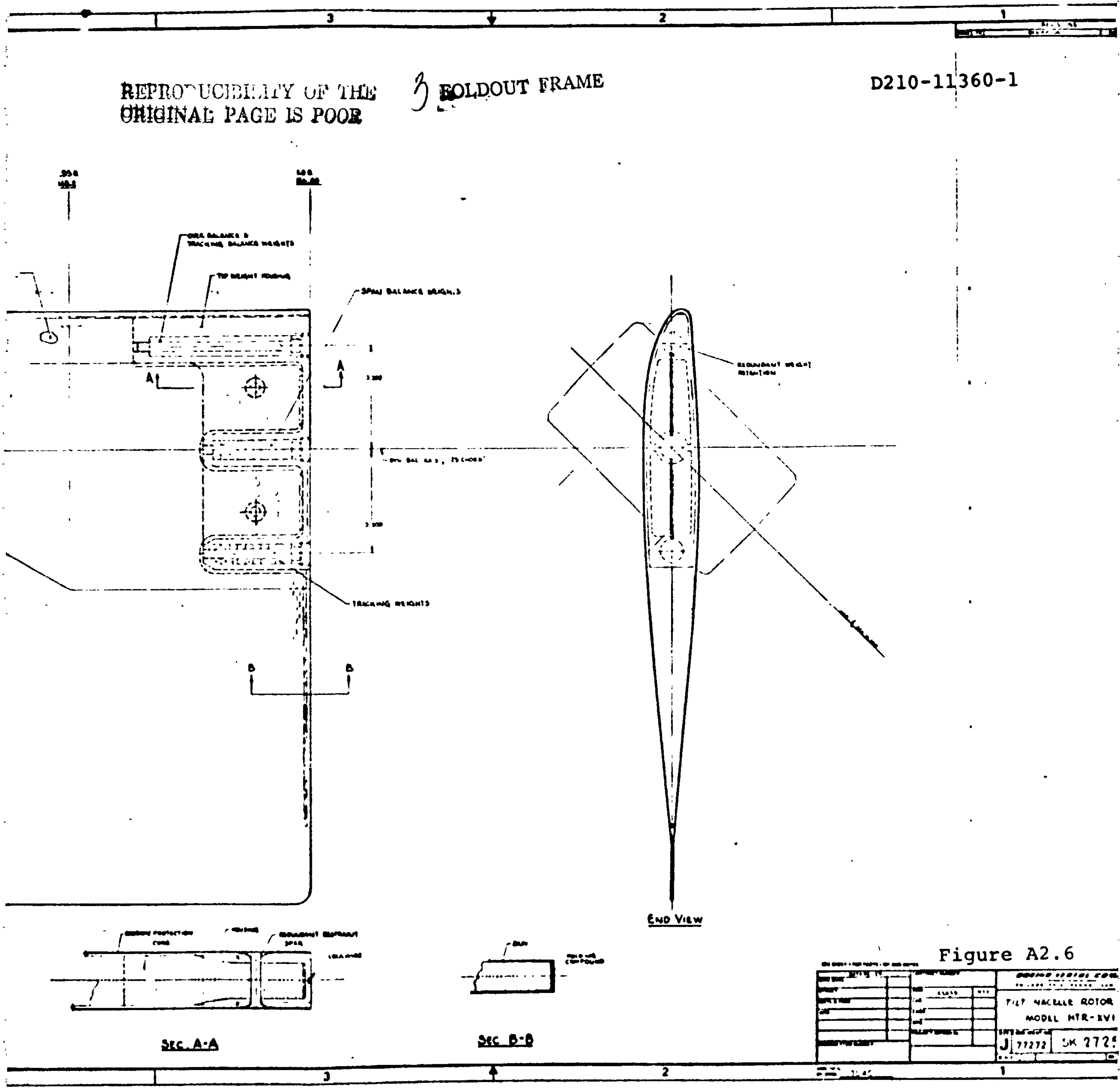

Figure A2.6

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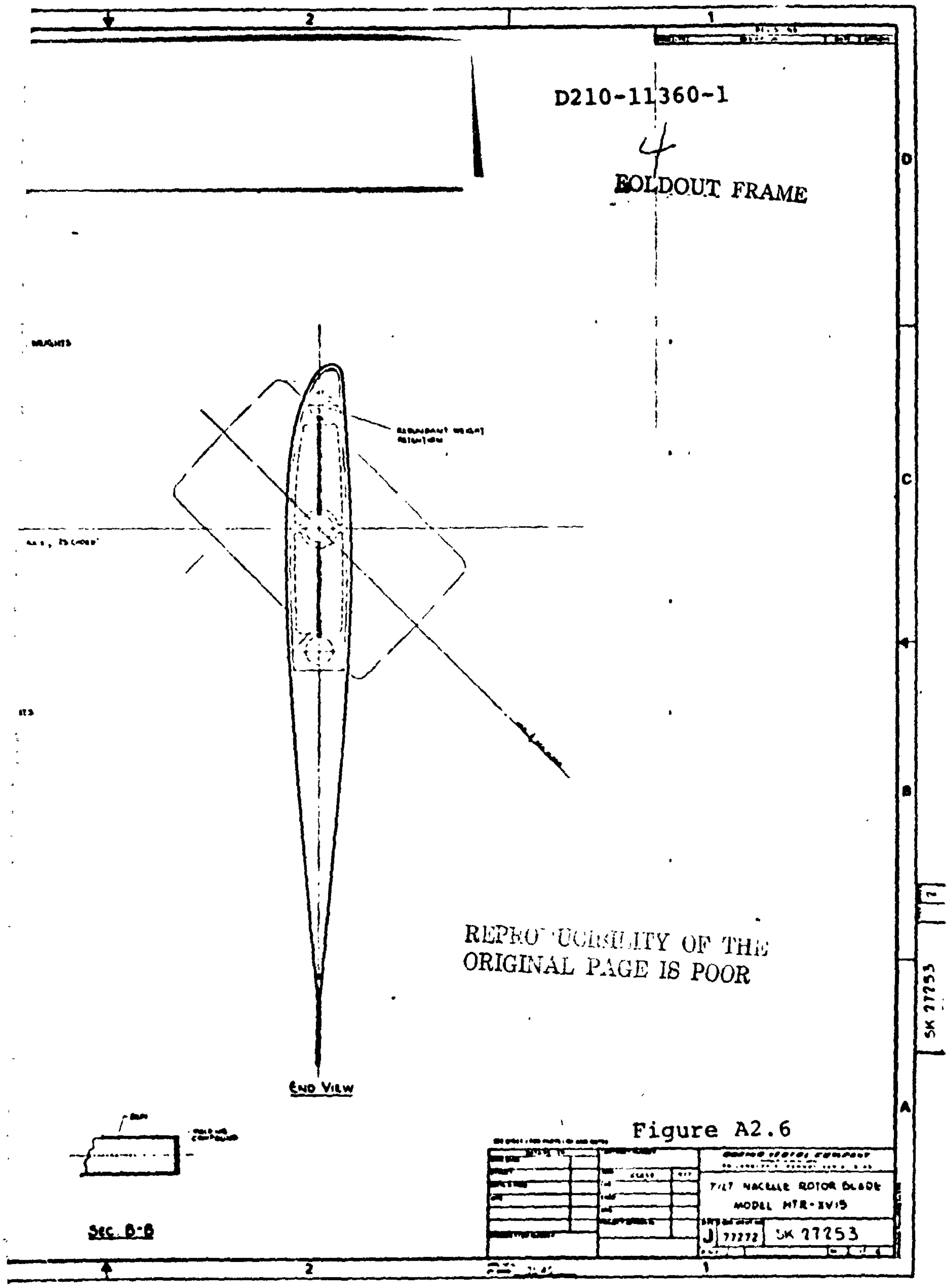

Figure A2.6

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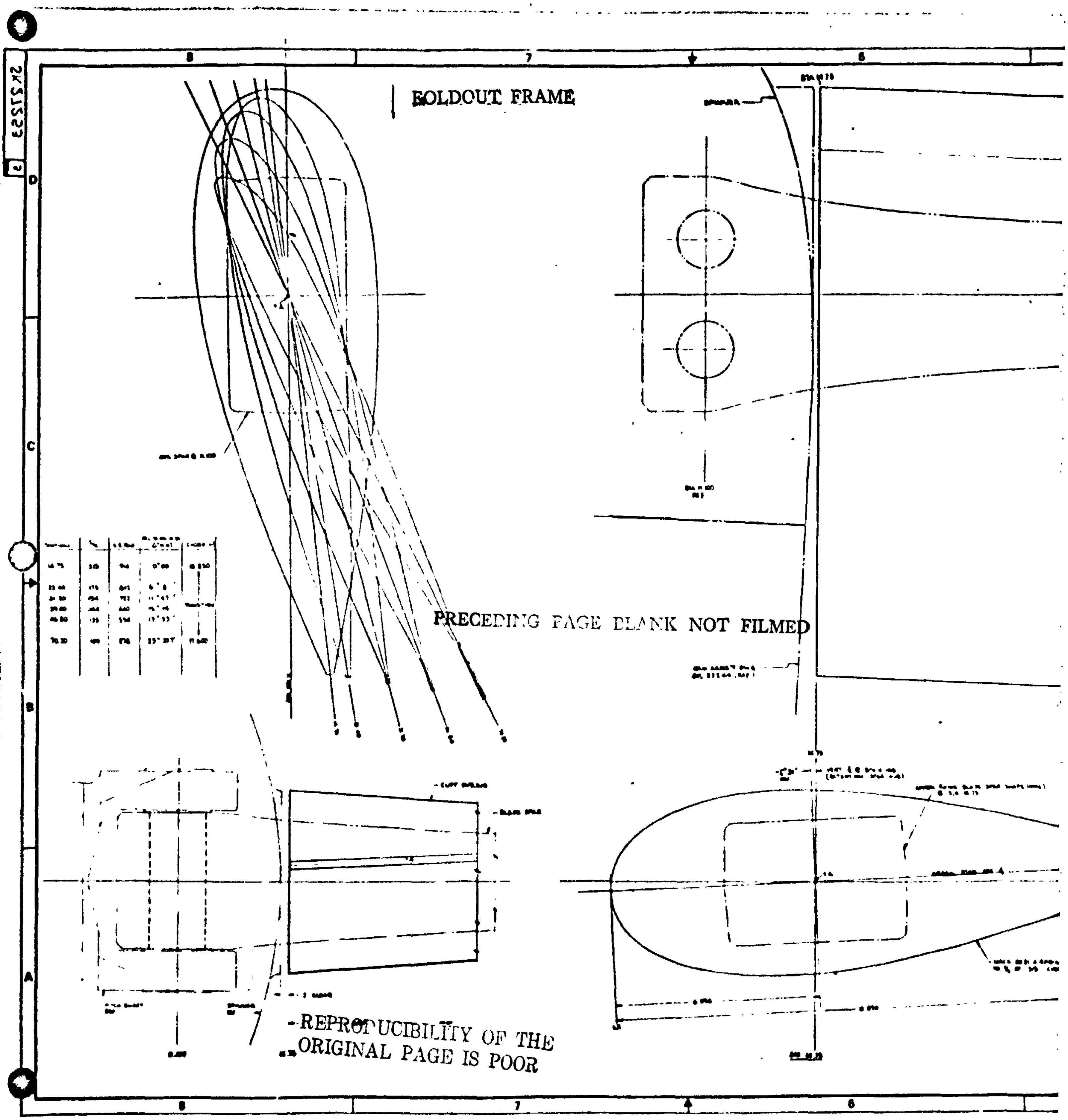

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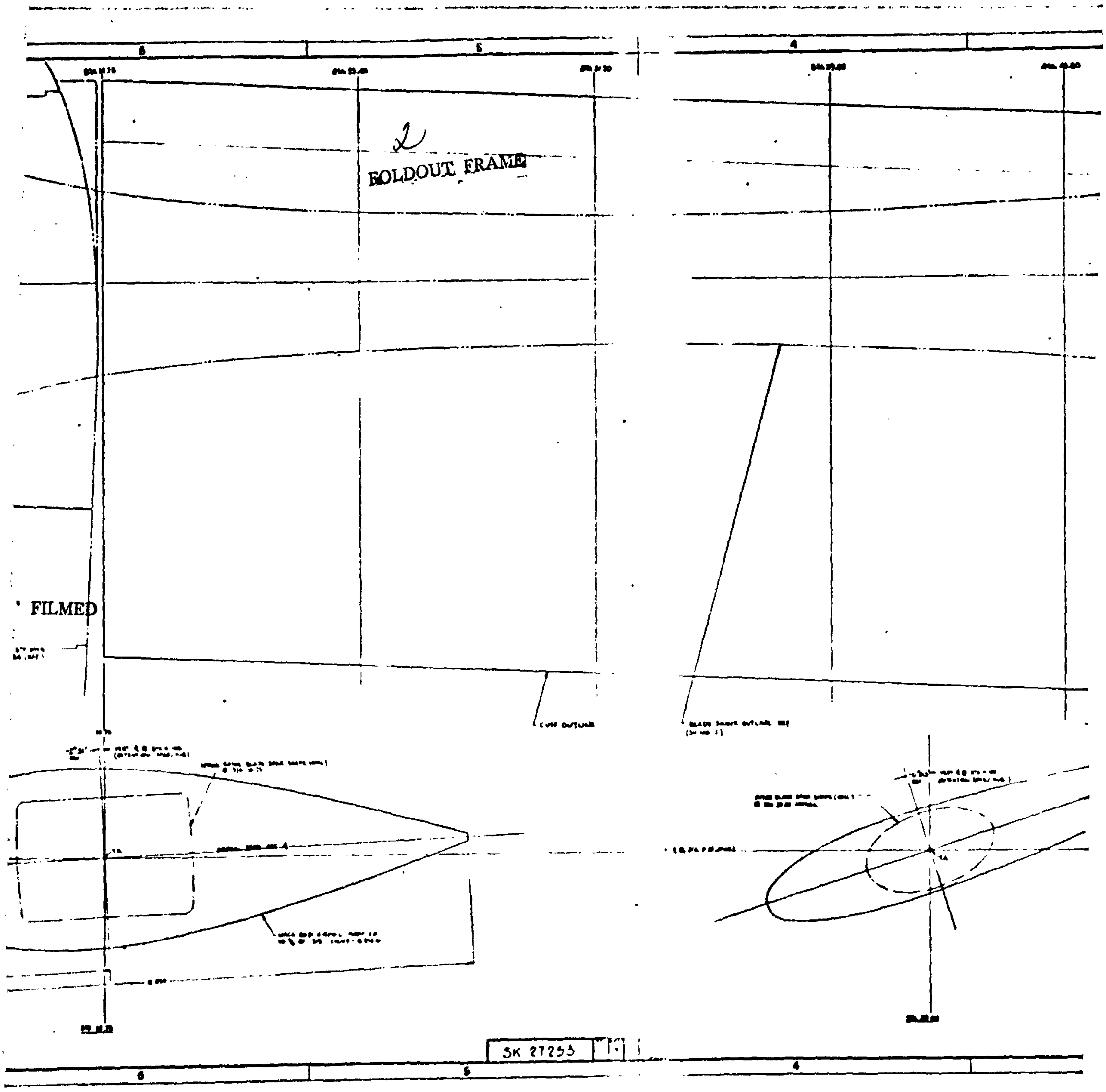

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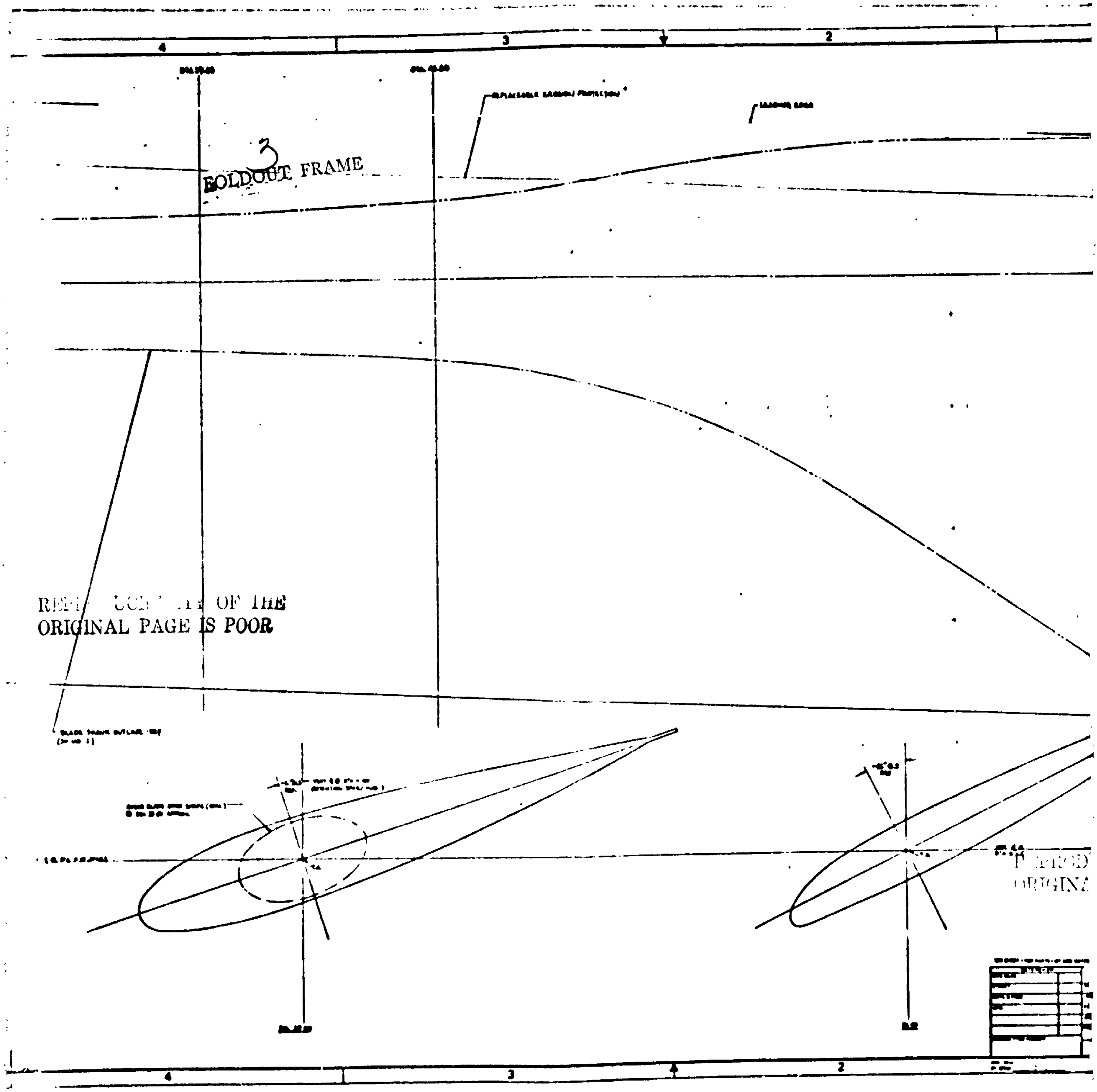

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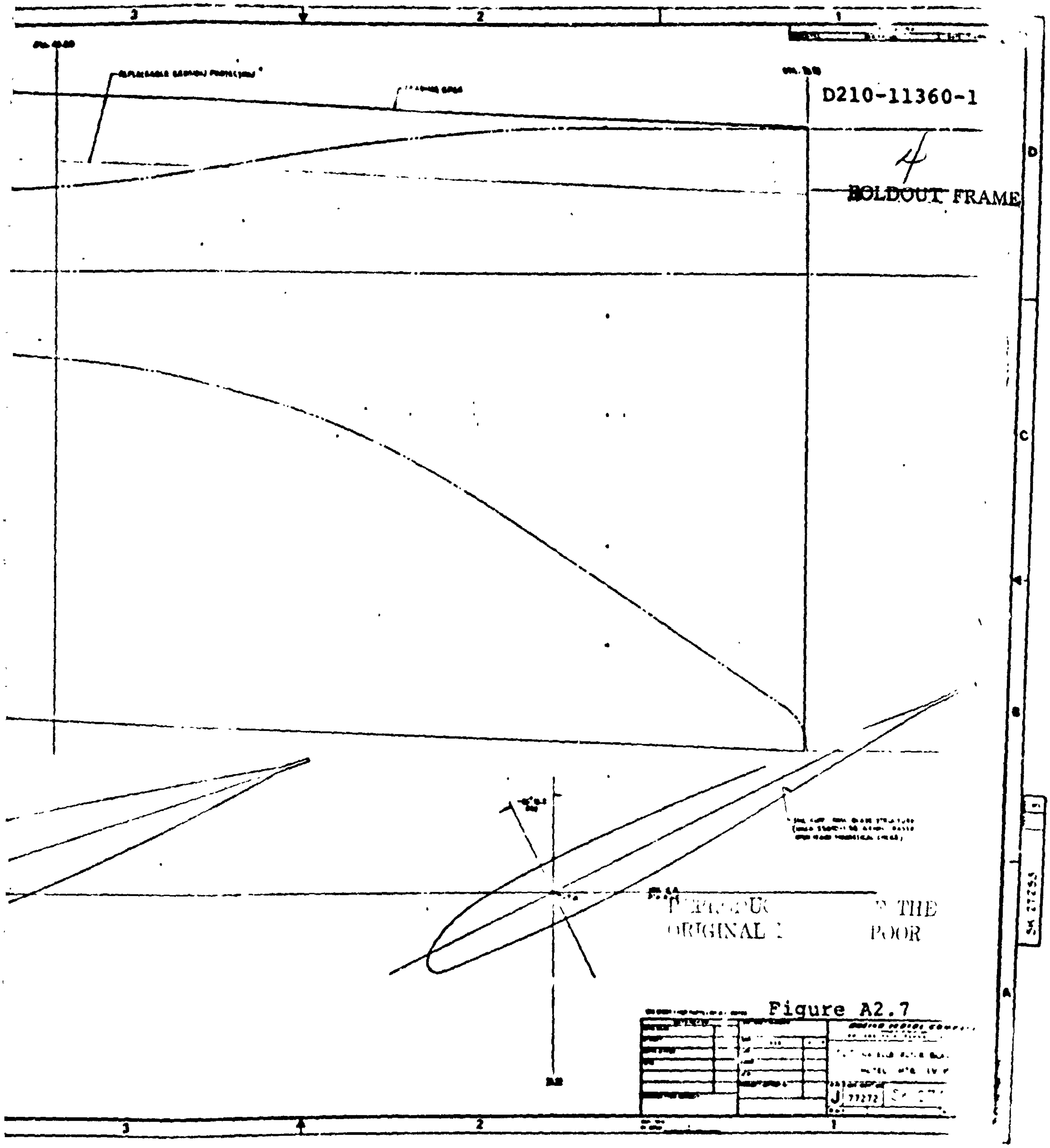


Figure A2.7

airfoil skins, inner torsion wrap and outer torsion wrap. The blade root end, a substitution was made by changing the boron outer torsion wrap to glass/graphite combination. "Overall" geometry, twist, chordal dimension, and blade radius remain as the Model 222 rotor blade configuration.

Blade/Hub Interface: The two-pin retention concept selected, interfaces with the rotor hub pitch shaft at rotor Station 11.100". The pins center spacing was chosen to be 3.500 inches after a trade study conducted to optimize the attachment area for minimum weight and overall minimum cross section. The blade (SK-27253) slips into a horizontal clevis joint of the hub (SK-27252). Two vertical-cylindrical hollow retention pins connect the rotor blade to hub. Anti-fretting material is bonded to the faying surfaces of the rotor blade and is replaceable.

Structural Design

(A) Spar - The S-glass (SP-250 SF-1) uni-fibers, which form the primary load path, extend inboard from the blade tip around the two attachment loops and return outboard of the tip. This forms a dual continuous, bond-free, fail-safe blade root end retention. The thickness varies with the structural strength requirement. Graphite cross-ply has been added to the glass cross-plied skins for additional torsional strength.

The vertical wraparound root loops, transition to an elliptical inboard spar section whose characteristics are tailored to locate the "effective hinge" of the hingeless rotor system, as far inboard as possible. This section transitions to a "C" spar section at the inboard end of the airfoil portion of the blade. The "C" shape of the spar extends to the tip of the blade.

(B) Aft Fairing - The aft fairing consists of S-glass $\pm 45°$ cross-plied skins, over a Graphite $\pm 45°$ sub-strate, covering a Nomex honeycomb core. The inboard termination of the skins is at an angle to the spar to transfer the fairing loads onto the spar. The forward portion of the core, upper and lower surfaces are in contact with the cross-plied, S-glass inner torsion wrap, and the uni-directional spar material (the aft portion of the "C" spar).

The aluminum honeycomb core is continuous from inboard end of the airfoil to the blade tip. Aluminum core was selected for its higher shear modulus. And a more aft shear center requirement was met by the use of aluminum. Matched metal molds provide accurate control of the airfoil and T.E. angle. The T.E. of the fairing is a uni-directional S-glass build-up between upper and lower fairing skins. Further study should be made to optimize the core density at the forward, outer blade span.

Weight saving could result from this effort.

Sections: The entire lifting surface of the blade is contoured to NACA-23010-1.58 airfoil ordinates. The theoretical chord diameter is 17.600". The pitch axis, (twist axis also), is located .25 chord.

The actual chordal diameter is 18.850". Trim tab at T.E. increases this diameter by .75" locally. Spar sections are blended inboard to a near elipse shape at Station 31.20. Flaring and bending inboard to a rectangular shape at retententior. Station 11.100".

Instrumentation Provisions: Adequate strain-gaging and telemetry monitoring of the rotor blade is scheduled. The flap and chord bending strain gages shall be instrumented and afixed to the rotor blade shank and airfoil sections. Gauge placement shall be shown on an instrumentation installation drawing, due later. Judicious gauge placement shall monitor the critical shank sections and the maximum bending locations on the airfoil. Trailing edge readings will be recorded also.

Water Tightness: The epoxy resin/fiberglass matrix used throughout this rotor blade is an excellent moisture barrier. The inboard portion of the blade shank (at the retention holes) shall be sealed to assure a water-tight structure. The aft fairing inboard rib is conformed to bond over the fairing skins to prohibit the water entry path into the fairing core. At the tip (the only other possible opening in the blade) a BMS 5-44 polysulphide rib is inset between the fairing skins to allow the moisture to pass over the fairing skins and not contact the "end rib". Water-tight seal conditions exist at all openings into the tip weight housing, thus a completely sealed rotor blade.

Erosion Protection: The leading edge of the rotor blade is protected by a "Three-Part System". The outboard 15% of the blade (Station 132.60 to tip), a replaceable nickel cap is proposed. Inboard of this, a 6AL-4V titanium sheet covering 10% of chord, continues inboard to approximately Station 74.00. The leading edge of the aerodynamic fairing will be covered with a non-metallic sheeting - bonded secondarily and more easily replaced. Candidates for this covering are Dunlop's adhesive backed polyurethane sheet, WX1119 or Goodrich's Estane No. 58370-288 Black. The titanium sheet LE cap selected for minimum strain incompatibility with the fiberglass sub-strate. Nickel was selected for best wear characteristics in the highest wear rate region, and forms the LE tip cap.

Balance: Chordal balance is accomplished by segmented tungsten L.E. weights installed at the major assembly/bond. This weight system extends spanwise along the blade from Station 72.00 to 150.25. Selective assembly of blade components shall reduce the total L.E. balance weight required.

Tungsten tracking weights are contained in the tip-weight housing and are retained by a standard screw and a redundant, separate, threaded retainer which is aerodynamically smooth at the blade tip. These retainers are safety-wired after installation. The tracking weight product moment available (about the dynamic balance axis) is 3.13 inch-lbs. An additional .50 lb weight pocket on the "D.B.A." is additional span weight. Overbalance considerations are satisfied by the cavity available at the forward tracking weight location. The overbalance weights are installed prior to the forward tracking weights. If no overbalance, or tracking weight is required, a wood dowel is used to fill the unused cavity, similarly for other unused cavities.

Frequency Tuning: Some flap frequency tuning can be accomplished by the addition of weights at Station 124.80 (.80r). Three cavitites in the core assembly have a silicone/tungsten mass placed on an anti-mode to raise the second flapping frequency. Additionally, the tip fitting has a pocket capable of holding ten (10) weights, totaling .47 lbs; (at location Station 154.0). This mass lowers the first mode flapping frequency.

Twist: Blade twist remains much the same as the previous built design, with a slight deviation at the inboard end. In order to accommodate the two-pin retention assembly, the blade/spar twist begins outboard of the hub clevis hardware Station 13.26 or .085r. Inboard of this location through the clevis no twist exists. For twist chart data, see Sheet 1 of Drawing SK-27253, Zone 12A.

t/c: A 10% thickness of a theoretical chord of 17.600 = 1.760". An actual chord diameter is 18.850, a t/c of .093. The most inboard station of the aerodynamic fairing is Station 14.75 and is a modified NACA 0021 airfoil. Its t/c = 5.938/18.850 = .315. This shape is transitioned to NACA 23010-1.58 at Station 70.20".

Repairability: Fiberglass rotor blades are more easily repaired than a metal counterpart. The more forgiving fiberglass also reduces the number of repairs required. These repairs are usually made "in the field", rarely requiring a depot fix. Most repair requirements occur to the fairings and trailing edges. Temporary repair can be made by "taping over" the damage and continue flight operations until a convenient time to make a permanent repair.

A permanent repair involves the removal of a portion of the fairing, by machining out a section of the skin and core and the installing and bonding of a "patch". A retracking of the blade is then done dynamically on the aircraft.

Ice Protection - Eliminated from the requirements criteria.

Lightning - Eliminated from the requirements criteria.

Corrosion Control - Fiberglass non-metallic sleeves - CRES hardware. Paint against ultra-violet rays. Sealed end closures on airfoil portion.

AII-4 - SECTION PROPERTIES

For this design effort, the blade was divided into two major parts, the root end from Station 11.1 to Station 46.8, and the airfoil section at Station 70.2.

Seven root end iterations were required to meet the frequency and strength criteria. The final section properties of Design 37 are shown in Table A2.2. These values were calculated from the root end cross sections shown on SK-27253.

The influence of the blade-to-hub attachment joint and the pitch change bearings on stiffness has been determined. The flap and chord stiffnesses from .017R to .09R are shown in Table A2.2. The stiffness calculations include a 3 inch section of the blade outboard of .071R, the fiberglass loops around the pin, the pin, the pitch shaft outboard of the pins, the remaining portion of the pitch shaft, the hub arm and the pitch change bearings. These calculations were performed for the titanium hub configuration and the root end configuration of Design 36.

For the airfoil section at .45R the objective was to minimize the weight and maximize the torsional stiffness when the boron crossply was removed. The iteration designated Design H met these objectives. A comparison between the properties of Design H and the "as manufactured" Model 222 is shown in Table A2.3. The cross section at .45R is shown on SK-27253.

Cuff: A cuff (inboard aerodynamic fairing - Station 14.75 to Station 70.20) will be separately mounted/bonded in place. The composite cuff shall be structurally independent of the rotor blade shank. It shall have a leading edge erosion protection which is replaceable. A candidate material for this application is Goodrich's "Estane 58370", (a polyurethane strip). Instrumentation wiring provisions shall be provided to complete connections between blade shank sensors/gauges and the hub recording device terminal board; and or telemetry slipring assembly mounted on the hub.

The cuff planform is shifted forward at the inboard end (Station 14.75) to more easily encapsulate the rotor blade shank. The cuff (entotal) is to be a removeable item as easy access to the blade shank, for periodic inspection.

Cuff weight estimates are 18 to 20 pounds, and are included in a total blade weight of approximately 93 pounds.

TABLE A2.2
HTR-XV15 ROOT END
SUMMARY OF SECTION PROPERTIES AND FATIGUE MARGINS

DESIGN 37

r/R	WT LB/IN (1)	AE LB x 10^{-6}	EI_F LB-IN^2 x 10^{-6}	EI_C LB-IN^2 x 10^{-6}	GJ LB-IN^2 x 10^{-6}	I_p LB IN^2/IN (2)	FATIGUE M.S. (3)
0	7.2		1 x 10	1 x 10			
.017	7.2		1 x 10	1 x 10			
.017	7.2		580.8	680.7			
.071	7.2		580.8	680.7			
.071	.962	55.8	580.8	680.7	87.8	2.13	+.10
.090			580.8	680.7	87.8		
.10	.688	47.0	72.7	144.8	53.4	1.54	+.88
.15	.438	30.7	22.7	46.6	22.8	.91	+.10
.20	.349	24.4	12.0	31.9	15.4	.97	+.03
.25	.283	18.4	9.5	29.1	15.8	.80	+.01
.30	.251	14.5	7.1	30.8	15.6	.84	-.04
.40	.456	12.36	4.96	495.0	11.3	12.7	
.50	.442	11.4	4.38	475.0	11.3	12.7	
.60	.434	10.3	3.77	465.0	11.0	12.7	
.70	.429	10.1	3.67	470.0	10.8	12.62	
.75	.426	10.0	3.48	470.0	11.8	12.60	
.80	.425	9.8	3.30	475.0	12.8	13.59	
.85	.424	9.9	3.31	482.0	14.0	12.53	
.90	.415	10.1	3.32	490.0	15.3	12.51	
.95	.415	10.0	3.29	490.0	15.7	12.51	
.975	.415	10.0	3.26	490.0	15.9	12.51	
.975	.415	10.0	3.26	490.0	15.9	12.51	
1.0	.415	9.9	3.24	490.0	16.2	12.51	

(1) BALANCE WEIGHTS NOT INCLUDED IN LB/IN. TABULATION. THESE ARE 1.33 LBS AT .7725, .80 AND .8225 r/R (i.e., NOMINALLY 4 LBS AT .80 r/R). TIP WEIGHT IS 4.2 LBS SPREAD OVER .9775 TO 1.0 r/R.

(2) VALUES FROM DESIGN 29A; VALUES FOR DESIGN 37 WILL NOT BE SIGNIFICANTLY DIFFERENT.

(3) COMPOSITE FATIGUE M.S. WITHOUT SHEAR STRAINS. EFFECT WAS JUDGED TO BE INSIGNIFICANT BASED ON CHECK MADE DURING DESIGN 31 AT r/R = .2 AND .25. FIGURES FROM 0.30 TO TIP NOT AVAILABLE.

TABLE A2.3

HTR-XV15 AIRFOIL SECTION
COMPARISON OF SECTION PROPERTIES
AT 45% R

	MODEL 222 ①	DESIGN H
WT, LB/IN	.436	.456
AE, LB X 10^6	13.34	12.36
EI_F, LB-IN^2 X 10^6	5.5	4.96
EI_C, LB-IN^2 X 10^6	540	495
GJ, LB-IN^2 X 10^6	20.54	11.3
I_p, LB-IN^2/IN	12.69	12.75
CG_c, % C	25	25
NA_c, % C	33.1	30.7
SHEAR CENTER, % C	23	22

① AS MANUFACTURED BLADE,

C = 18.85 INCHES

The blade weight for the final root end and airfoil designs has been determined. A weight summary is shown in Table A2.4. The blade weight is 93 pounds. The total rotor system weight is 501 pounds. This compares to the Model 222 total rotor system weight of 516 pounds.

<u>Blade Frequencies</u>: Using the Design 37 root end and the Design H airfoil properties the Y-71 coupled natural frequencies were determined. For the hover condition (551 rpm and 9° collective pitch) all the frequencies meet the design criteria as shown in Table A2.5.

Table A2.6 shows the variation of natural frequencies with rpm and collective pitch. If changes are made in the final design the sensitivity of frequencies to tip weight and tuning weight at .8R is presented in Table A2.7 for the hover condition.

<u>Design Loads</u>: This section contains the fatigue loads used for this design effort. Table A2.8 shows the alternating flap, chord and torsional moment distributions and Figure A2.9 presents the steady moment distributions. Figure A2.10 shows the design CF distribution at 551 rpm.

The magnitude of the alternating flap and chord moments was selected to give infinite life for the metal hub components and a greater than 5,000 hours life for the fiberglass spar. In terms of hover cyclic pitch, the metal components will be designed for ±7° and the fiberglass spar will be designed for ±5°. The distribution of alternating flap and chord moments was obtained from Reference 3 in Section 5 for

TABLE III

HTR-XV15

WEIGHT SUMMARY FOR DESIGN 37

SECTION/PART	WEIGHT
r/R = .25 - 1.0	55.31 (1)
r/R = .071 - .25	11.98
GLASS & GRAPHITE (INBOARD OF .071R)	1.56
2 SLEEVES	.86
4 DROOP STOPS	1.44
STRUCTURAL FOAM	1.06
BULKHEAD	.75
CUFF (EST.)	20.0
	Σ92.96 LBS/BLADE

(1) WEIGHT BASED ON MODEL 22 "AS MANUFACTURED" WEIGHT DISTRIBUTION PLUS ADDITIONS DUE TO AIRFOIL DESIGN H. ALSO INCLUDES 4.2 POUNDS TUNING WEIGHT AT .8R AND 4.0 POUNDS TIP WEIGHT.

TOTAL SYSTEM WEIGHT

HUB	222.2 LB
3 BLADES	278.9 LB
	Σ501.1 LB

TABLE A2.4. HTR-XV15 WEIGHT SUMMARY FOR DESIGN 37

HTR-XV15

COMPARISON OF Y-71 NATURAL FREQUENCIES

551 RPM, 9° COLLECTIVE

TABLE IV

ω/Ω	MODE	CRITERIA ①	DESIGN 37 ③	AS MANUFACTURED MODEL 222 ②
1ST	CHORD	.72 - .76	.747	.696
2ND	FLAP	1.15 - 1.25	1.219	1.201
3RD	FLAP	3.2 - 3.8	3.232	3.134
4TH	TORSION	3.65 - 3.75	3.665	4.288
5TH	FLAP	---	5.797	5.708

① REFERENCE - IOM 8-7040-1-729, HTR-XV15 ROTOR SYSTEM DESIGN CRITERIA.

② WITH MINIMUM TIP WEIGHT (2.69 LB).

③ WITH 4.2 LBS AT .8R AND 4 LBS AT TIP.

TABLE A2.5. COMPARISON OF Y-71 NATURAL FREQUENCIES, 551 RPM, 9° COLLECTIVE

HTR-XV15

Y-71 NATURAL FREQUENCIES VS RPM AND

COLLECTIVE PITCH

ROTOR RPM	MODE	COLLECTIVE PITCH 9°	25°	40°	60°
125	1	1.551 F	1.463 F	1.376 F	1.291 F
	2	2.642 C	2.691 C	2.735 C	2.776 C
	3	6.758 F	6.744 F	6.726 F	6.702 F
	4	15.62 T	15.61 T	15.59 T	15.57 T
	5	16.29 F	16.28 F	16.28 F	16.27 F
250	1	1.154 C	1.043 C	.953 F	.876 F
	2	1.534 F	1.611 F	1.664 C	1.705 C
	3	4.238 F	4.216 F	4.186 F	4.148 F
	4	7.854 T	7.834 T	7.805 T	7.763 T
	5	9.101 F	9.092 F	9.080 F	9.066 F
386	1	.909 C	.837 C	.778 C	.731 F
	2	1.299 F	1.346 F	1.380 F	1.405 C
	3	3.540 F	3.513 F	3.477 F	3.431 F
	4	5.139 T	5.110 T	5.065 T	4.999 T
	5	6.870 F	6.857 F	6.843 F	6.825 F
551	1	.747 C	.706 C	.672 C	.647 F
	2	1.219 F	1.242 F	1.260 F	1.273 C
	3	3.232 F	3.202 F	3.163 F	3.112 F
	4	3.665 T	3.623 T	3.559 T	3.466 T
	5	5.797 F	5.783 F	5.767 F	5.749 F
750	1	.648 C	.625 C	.607 C	.595 F
	2	1.180 F	1.192 F	1.201 F	1.206 C
	3	2.771 T	2.715 T	2.629 T	2.501 T
	4	3.072 F	3.040 F	2.998 F	2.945 F
	5	---	---	---	---

DOMINANT MODE: F - FLAPWISE
C - CHORDWISE
T - TORSIONAL

TABLE A2.6. Y-71 NATURAL FREQUENCIES VS RPM AND COLLECTIVE PITCH - DESIGN 37

HTR-XV15

EFFECT OF TIP WEIGHT AND TUNING WEIGHT AT .8R

DESIGN 37

551 RPM, 9° COLLECTIVE

	TIP WEIGHT - LBS ①				TUNING WT. AT .8R - LBS ②			
ω/Ω	2	3	4 ③	5	2.2	3.2	4.2 ③	5.2
1ST	.767	.756	.747	.738	.756	.751	.747	.742
2ND	1.229	1.224	1.219	1.215	1.223	1.221	1.219	1.217
3RD	3.219	3.223	3.232	3.245	3.214	3.223	3.232	3.240
4TH	3.665	3.665	3.665	3.665	3.665	3.665	3.665	3.665
5TH	5.763	5.775	5.797	5,825	5.852	5.822	5.797	5.774

① TUNING WEIGHT AT .8% = 4.2 LBS.

② TIP WEIGHT = 4 LBS.

③ BASELINE.

TABLE A2.7. EFFECT OF TIP WEIGHT AND TUNING WEIGHT AT .8R

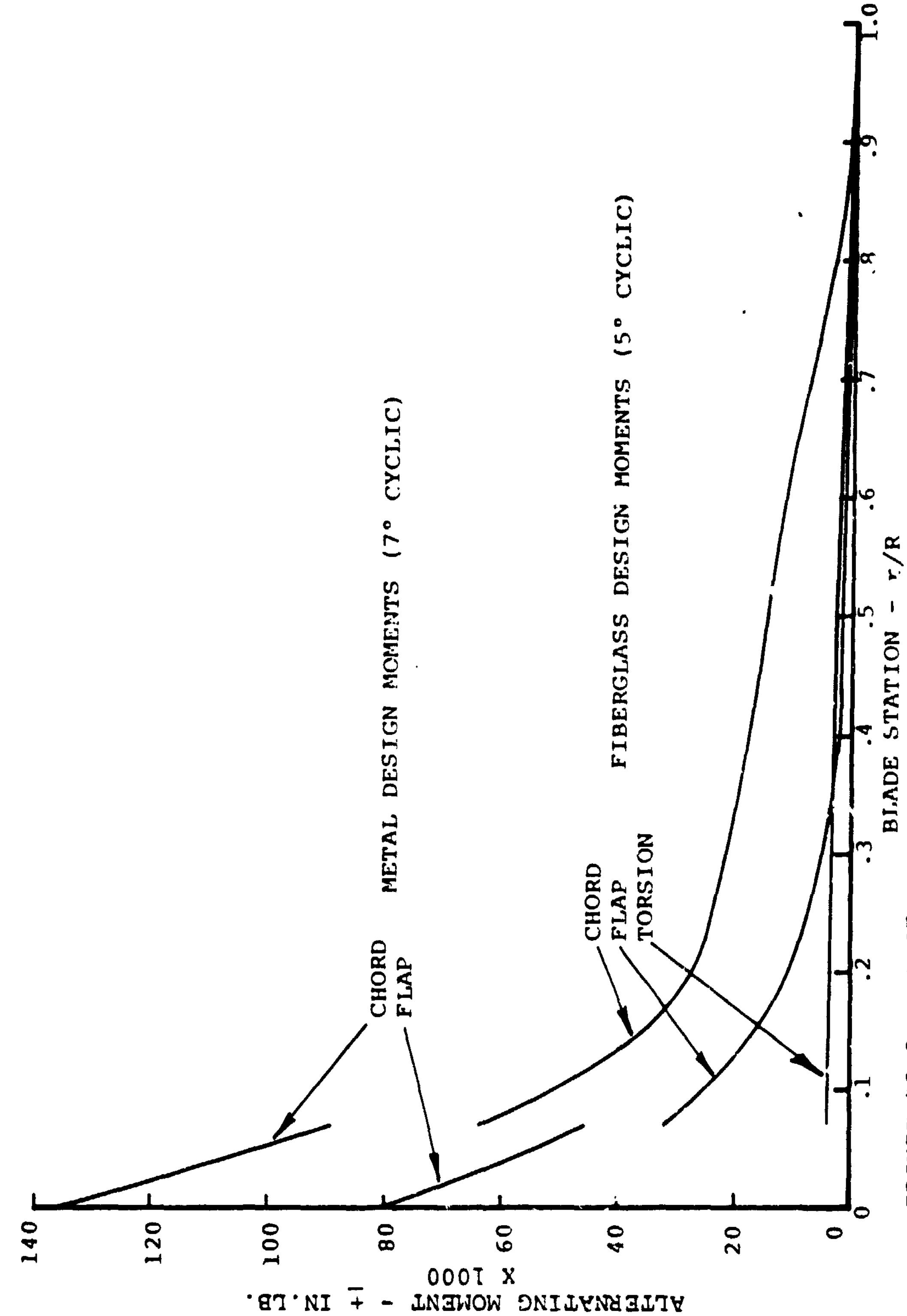


FIGURE A2.8. XV-15 HINGELESS TILT ROTOR BLADE - DESIGN ALTERNATING MOMENTS

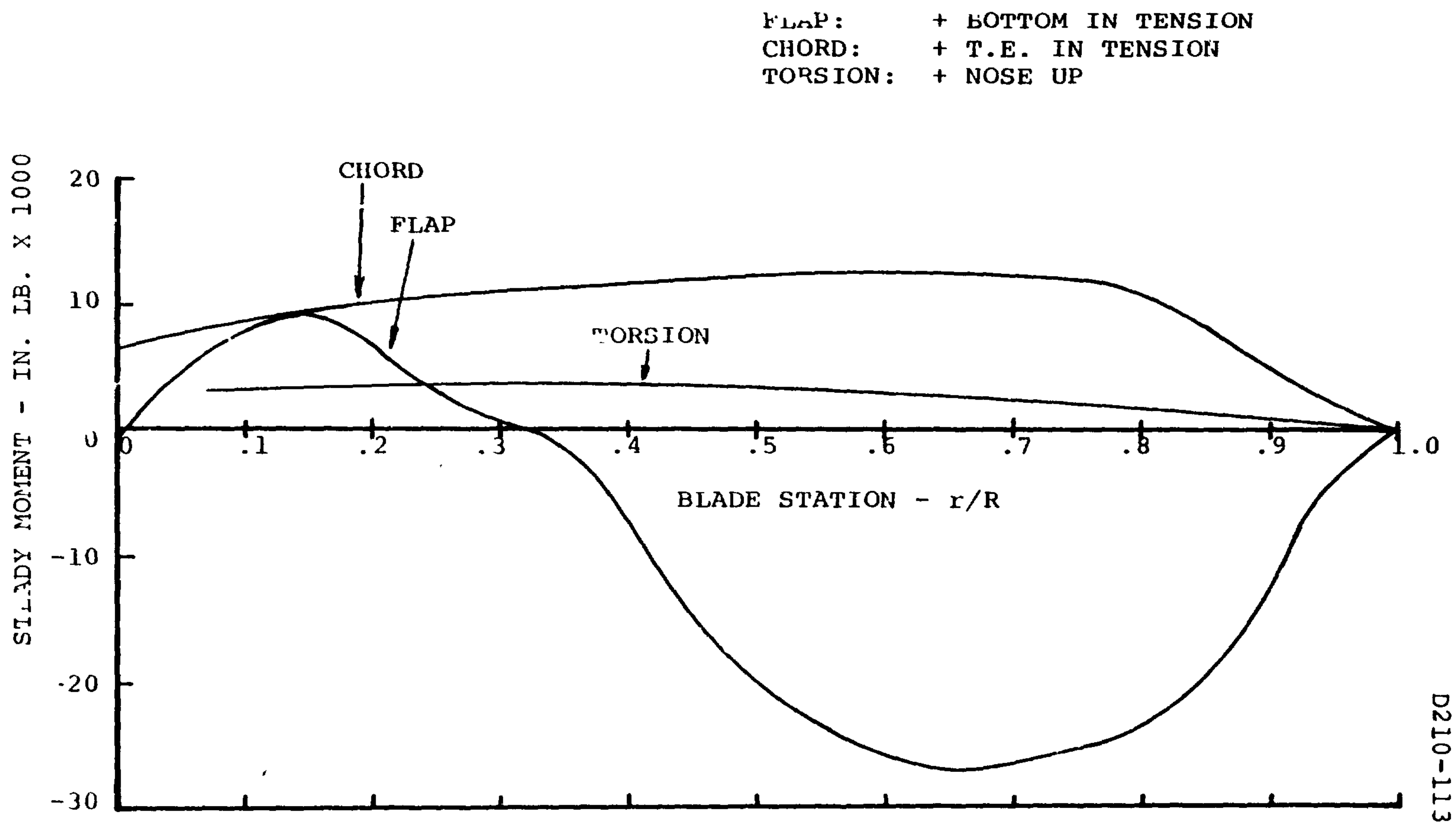


FIGURE A2.9. XV-15 HINGELESS TILT ROTOR BLADE - DESIGN STEADY MOMENTS.

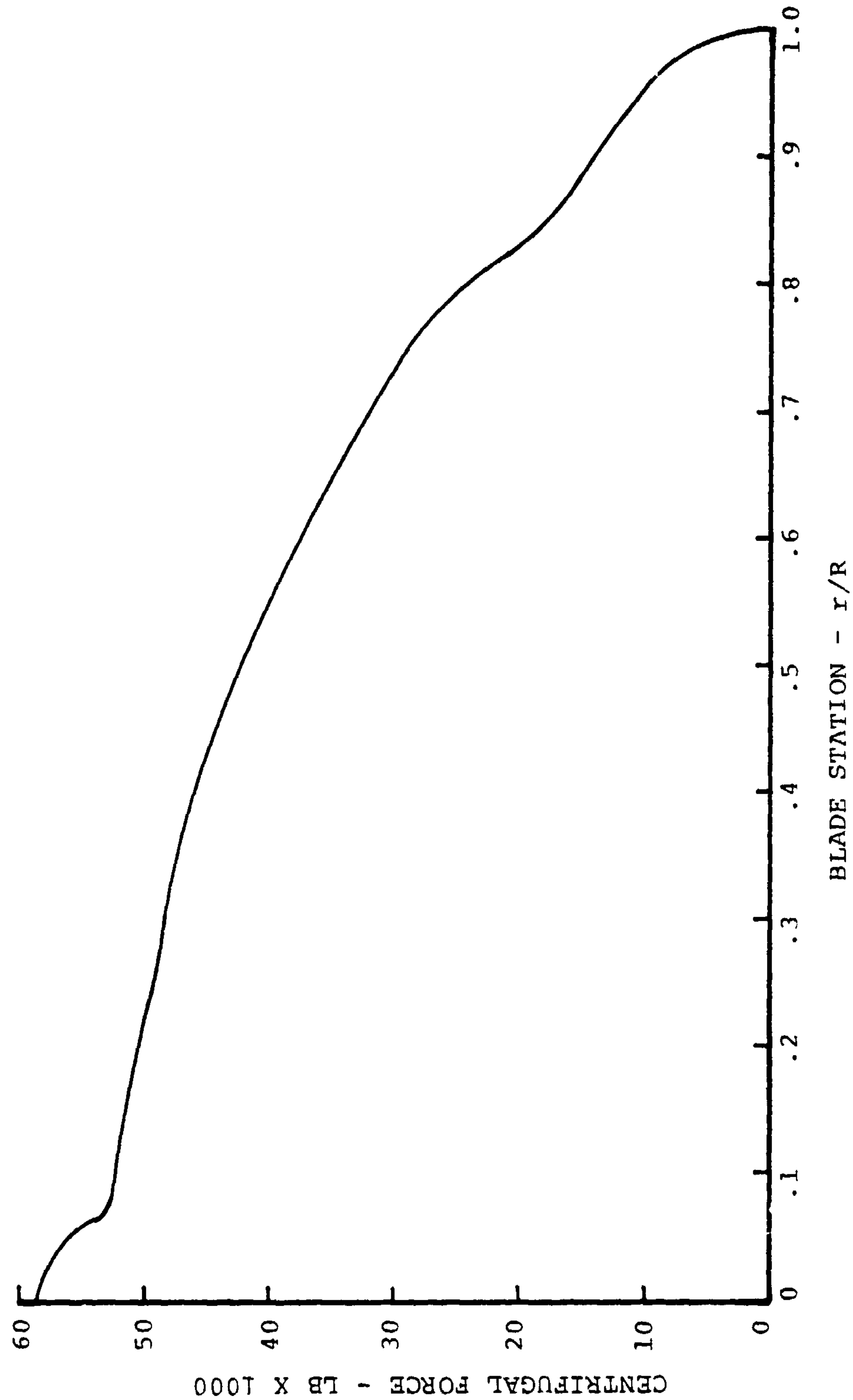


FIGURE A2.10. XV-15 HINGELESS TILT ROTOR BLADE - DESIGN OF DISTRIBUTION - 551 RPM the hover flight mode. The alternating torsional moment at the pitch link attachment was determined by multiplying the maximum measured pitch link load of ±1,000 pounds (Reference 3 in Section 5) and the 4-inch pitch arm offset. The spanwise distribution of torsional moment was obtained from Reference 5 in Section 5.

The steady design moments are representative of the steady loads at the hover design condition (551 rpm and 9° collective pitch). The steady bending moments at 12.4% R were obtained from the 1/4 scale wind tunnel model (Reference 6 in Section 5). The design torsional moments are based on a measured pitch link load of -900 pounds (Reference 3 in Section 5 for 551 rpm and 9° collective pitch. The spanwise distribution was calculated using the methodology contained in Reference 5 in Section 5.

Figure A2.10 presents the design CF distribution at 551 rpm. The weight distribution was composed of two parts. From 45%R to the tip the weight of the Model 222 "as manufactured" blade was used. This distribution also includes provisions for 4 pounds of tip weights and 4.2 pounds of tuning weights at 80%R. From 7.1%R, the blade attachment joint, to 30%R the weight distribution of Design 31 was used.

<u>Fatigue Analysis</u>: Using the design loads from Section 4.0 a fatigue analysis of the Design 37 root end was performed. The details are presented on the following pages. The tension fatigue allowables are shown in Figure A2.11.

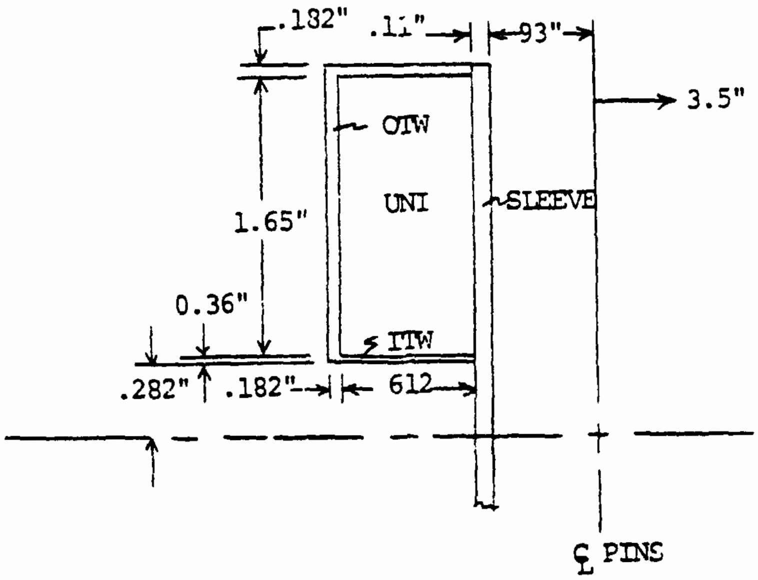
HTR-XV15

FATIGUE STRAIN ANALYSIS

OF FIBERGLASS AT r/R = .071

3.5 INCH PIN SPACING
PIN O.D. = 1.85" + .01" FRETTING PROTECTION
SLEEVE THICKNESS = .1" + .01" ADHESIVE
SPAR THICKNESS = 4.34" (INCLUDES .04" FIBERGLIDE)
PACK HEIGHT = 1.65"

OTW:	.036"	± 45°	SP250/SF1
	.11"	∓ 45°	GRAPHITE
	.036"	∓ 45°	SP250/SF1
UNI:	.612"	0°	SP250/SF1
ITW:	.036"	± 45°	SP250/SF1



$$AE_{OTW} = (1.78)(.036)(2)(.612 + .182 + 1.65) + (2.1)(.11)(.612 + .182 + 1.65)$$

$$= .878 \times 10^6$$

$$AE_{UNI} = (6.3)(1.65)(.612) = 6.362 \times 10^6$$

$$AE_{ITW} = (1.78)(.036)(.612 + .182) = .051 \times 10^6$$

$$AE/_{PACK} = 7.291 \times 10^6$$

STRESS CONCENTRATION ON UNI FIBERGLASS

$$\frac{r_o}{r_L} = \frac{.93 + .11 + .612}{.93 + .11} = 1.588$$

$$K_{t\ \mathrm{THEORY}} = \frac{1.3(1.588)^2 + .7}{1.588 + 1} = 1.538$$

$$K_{t\ \mathrm{TEST}} = (1.538)(1.332) = 2.047$$

LOAD/LOOP (LOADS REFERENCE FIGURES

$$P_{CF} = \frac{53000}{4} = 13{,}250 \text{ LB}$$

$$P_{MF} = \frac{6160 \pm 32500}{4(\frac{1.65}{2} + .036 + .282)} = 1{,}347 \pm 7{,}108 \text{ LBS}$$

$$P_{M_C} = \frac{8200 \pm 63300}{(2)\ (3.5)} = 1{,}171 \pm 9{,}043 \text{ LBS}$$

STRAINS/PACK

$$\varepsilon_{STDY} = \frac{(2.047)(13250 + 1347 + 1171)}{(2)(7.291 \times 10^6)} = 2{,}214\ \mu \text{ IN./IN.}$$

$$\pm\varepsilon = \frac{(2.047)(7108 + 9043)}{(2)\ (7.291 \times 10^6)} = \pm\ 2{,}267\ \mu \text{ IN./IN.}$$

$$\varepsilon_{TOTAL} = 2{,}214 \pm 2{,}267\ \mu \text{ IN./IN.}$$

$$\text{M.S.} = \frac{2490}{2267} - 1 = \underline{\underline{+.10}}$$

HTR-XV15

FATIGUE STRAIN ANALYSIS

OF ROOT END

DESIGN 37

LOADS ARE FROM FIGURES 5, 6 AND 7

DIMENSIONS ARE FROM SK-27253.

THE CHECK POINT AT EACH STATION IS FOR THE MAXIMUM ALT. STRAIN.

ALLOWABLES ARE FROM FIGURE 8

1) r/R = .1, STATION 15.6

$$\epsilon_{CF} = \frac{52300}{47 \times 10^6} = 1{,}113 \ \mu \text{ IN./IN.}$$

$$\epsilon_M = \frac{(7920)(1.67)}{72.7 \times 10^6} + \frac{(8800)(2.53)}{144.8 \times 10^6} = 336 \ \mu \text{ IN./IN.}$$

$$\pm\epsilon_M = \frac{(25000)(1.67)}{72.7 \times 10^6} + \frac{(51700)(2.53)}{144.8 \times 10^6} = \pm 1{,}478 \ \mu \text{ IN./IN.}$$

$$\epsilon_{TOTAL} = 1{,}449 \pm 1{,}478 \ \mu \text{ IN./IN.}$$

$$\text{M.S.} = \frac{2780}{1478} - 1 = \underline{\underline{+.88}}$$

2) r/R = .15, STATION 23.4

$$\epsilon_{CF} = \frac{51500}{30.7 \times 10^6} = 1{,}678 \ \mu \text{ IN./IN.}$$

$$\epsilon_M = \frac{(9240)(1.18)}{22.7 \times 10^6} + \frac{(9550)(1.74)}{46.6 \times 10^6} = 837 \ \mu \text{ IN./IN.}$$

$$\pm\epsilon_M = \frac{(16700)(1.18)}{22.7 \times 10^6} + \frac{(35000)(1.74)}{46.6 \times 10^6} = \pm 2{,}175 \ \mu \text{ IN./IN.}$$

$$\epsilon_{TOTAL} = 2{,}515 \pm 2{,}175 \ \mu \text{ IN./IN.}$$

$$\text{M.S.} = \frac{2390}{2175} - 1 = \underline{\underline{+.10}}$$

3) r/R = .20, STATION 31.2

$$\epsilon_{CF} = \frac{50400}{24.4 \times 10^6} = 2{,}066 \ \mu \text{ IN./IN.}$$

$$\epsilon_M = \frac{(6820)(.7)}{12 \times 10^6} + \frac{(10100)(1.77)}{31.9 \times 10^6} = 958 \ \mu \text{ IN./IN.}$$

$$\pm\epsilon_M = \frac{(10800)(.7)}{12 \times 10^6} + \frac{(27500)(1.77)}{31.9 \times 10^6} = \pm\ 2{,}156 \ \mu \text{ IN./IN.}$$

$$\epsilon_{TOTAL} = 3{,}024 \pm 2{,}156 \ \mu \text{ IN./IN.}$$

$$\text{M.S.} = \frac{2230}{2156} - 1 = \underline{\underline{+.03}}$$

4) r/R = .25, STATION 39

$$\epsilon_{CF} = \frac{49300}{18.4 \times 10^6} = 2{,}679 \ \mu \text{ IN./IN.}$$

$$\epsilon_M = \frac{(3260)(.5)}{9.5 \times 10^6} + \frac{(10580)(2.0)}{29.1 \times 10^6} = 899 \ \mu \text{ IN./IN.}$$

$$\pm\epsilon_M = \frac{(7500)(.5)}{9.5 \times 10^6} + \frac{(24200)(2.0)}{29.1 \times 10^6} = \pm\ 2{,}058 \ \mu \text{ IN./IN.}$$

$$\epsilon_{TOTAL} = 3{,}578 \pm 2{,}058 \ \mu \text{ IN./IN.}$$

$$\text{M.S.} = \frac{2070}{2058} - 1 = \underline{\underline{+.01}}$$

5) r/R = .3, STATION 46.8

$$\epsilon_{CF} = \frac{48400}{14.54} = 3{,}329 \ \mu \text{ IN./IN.}$$

$$\epsilon_M = \frac{(700)(.48)}{7.1 \times 10^6} + \frac{(11000)(2.28)}{30.8 \times 10^6} = 826 \ \mu \text{ IN./IN.}$$

$$\pm\epsilon_M = \frac{(5800)(.48)}{7.1 \times 10^6} + \frac{(21700)(2.28)}{30.8 \times 10^6} = \pm\ 1{,}998 \ \mu \text{ IN./IN.}$$

$$\epsilon_{TOTAL} = 4{,}191 \pm 1{,}998 \ \mu \text{ IN./IN.}$$

$$\text{M.S.} = \frac{1910}{1998} - 1 = \underline{\underline{-.04}}$$

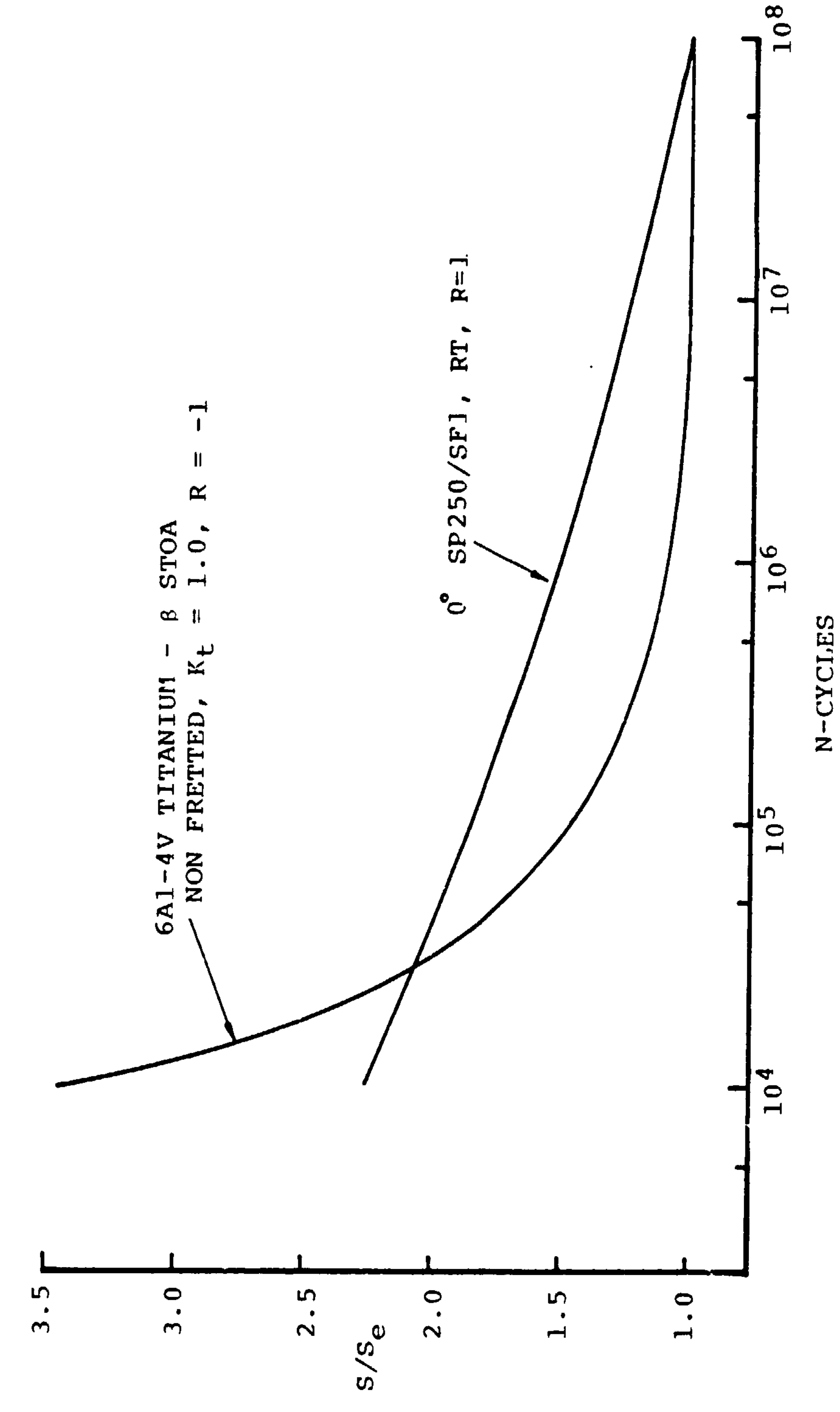


FIGURE A2.11. TENSION FATIGUE CURVE SHAPES

All the margins of safety at the check stations are positive except for Station 46.8 which is -.04. This resulted because of a late revision to the steady chord moment distribution. A one or two ply addition of uni-directional material at this station will remove the negative margin. The added material will have a small effect on the blade frequency and weight.

The fatigue analysis does not include the effect of shear strains. A check of shear strains was made on Design 31 at .2R and .25R. The shear strains lowered the margin by only 1%. As a result, they were not included in subsequent iterations.

Mission Profile and Fatigue Life: A mission profile has been developed for the HTR XV-15 based on the XV-15 Bell flight spectrum (Reference Bell Document 301-199-003). The mission profile is presented on Table A2.8, along with the alternating resultant moments at .125R for each condition. There are certain conditions in the profile for which moments have not been defined, but they comprise only 5.315% of the total time. These conditions are climb and descents in cruise, dive to Vc and recovery in cruise, power transitions and autorotation. The loads for these conditions should be developed in the detailed design phase.

Based on the loads in the mission profile, life calculations have been made for the blade root end and hub. Table A2.9 presents the details. The calculated life of the blade root end is 17,334 hours, while the hub has a life of 111,524 hours. These lives satisfy the objective of this preliminary design (Reference Section 4.0). It should be noted that these lives do not include the missing flight conditions mentioned above or the ground-air-ground damage that could occur. These items should be included in the detailed design phase.

HTR-XV15 MISSION PROFILE

I GROUND TAXI MANEUVERS - 13,000 POUNDS GW, 7.3% TOTAL TIME

A. ENGINE/ROTOR START AND SHUT DOWN

	RPM	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125R
	150	1.16	$.1000 \times 10^6$	5600
	280	1.16	$.1949 \times 10^6$	26900
	320	1.16	$.2227 \times 10^6$	13600
	400	1.16	$.2784 \times 10^6$	11600
	550	1.16	$.3828 \times 10^6$	12500
B. GROUND TAXI	551	1.5	$.496 \times 10^6$	24200

II. HELICOPTER FLIGHT - 39.4% TOTAL TIME

	C.G.	GROSS WEIGHT	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125R
A. TAKEOFF & LANDING	Aft	10500	.2	$.0661 \times 10^6$	6400
	Aft	13000	.8	$.2645 \times 10^6$	6400
	Aft	15500	.2	$.0661 \times 10^6$	6500
	Fwd	10500	.2	$.0661 \times 10^6$	19700
	Fwd	13000	.8	$.2645 \times 10^6$	24200
	Fwd	15500	.2	$.0661 \times 10^6$	27800
B. STEADY HOVER IGE & OGE	Aft	10500	.5	$.1653 \times 10^6$	4500
	Aft	13000	2.0	$.6612 \times 10^6$	4500
	Aft	15500	.5	$.1653 \times 10^6$	4500
	Fwd	10500	.5	$.1653 \times 10^6$	22600
	Fwd	13000	2.0	$.5612 \times 10^6$	28000
	Fwd	15500	.5	$.1653 \times 10^6$	33400

C. HOVERING MANEUVERS	% CONTROL	GROSS WEIGHT	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125R
	100	10500,	.08	$.0264 \times 10^6$	59400
	80	13000 &	.39	$.1296 \times 10^6$	48500
	60	15500	.78	$.2579 \times 10^6$	37500
	40		1.56	$.5157 \times 10^6$	26500

TABLE A2.8. HTR-XV-15 MISSION PROFILE

HTR-XV15 MISSION PROFILE (CONTINUED)

D. LEVEL FLIGHT, 13,000 POUNDS GW, FORDWARD & AFT C.G. [1]

V_{KTS}	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125 R AFT CG	± RESULTANT MOMENT AT .125 R FWD CG
35	2.5	$.8265 \times 10^6$	<6500	18500
40	1.0	$.3306 \times 10^6$	6500	21000
50	2.0	$.6612 \times 10^6$	11000	24500
60	3.0	$.9918 \times 10^6$	15100	27500
70	4.0	1.3224×10^6	18700	31000
80	3.0	$.9918 \times 10^6$	22000	34000
90	2.0	$.6612 \times 10^6$	25500	37500

E. TURNS, 13,000 POUNDS GW, FORWARD & AFT C.G. [1]

BANK ϕ°	V_{KTS}	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
15	40	1.0	$.3306 \times 10^6$	7000	21000
30	40	.7	$.2314 \times 10^6$	9500	25000
45	40	.3	$.0992 \times 10^6$	11500	30000
15	80	.4	$.1322 \times 10^6$	23500	37000
30	80	.3	$.0992 \times 10^6$	26000	41500
45	80	.2	$.0661 \times 10^6$	31000	49500
60	80	.1	$.0331 \times 10^6$	39000	63000

F. PULLUPS, 13,000 POUNDS GW, AFT CG

g's	V_{KTS}	% TIME	CYCLES/ 1000 HRS	AFT CG
2.0	40	.22	$.0727 \times 10^6$	10000
2.35	80	.22	$.0727 \times 10^6$	49400
2.0	40	.22	$.0727 \times 10^6$	7100
2.5	80	.22	.072[illegible] 10^6	40700
2.1	120	.22	.07[illegible]	46000

G. ACCELERATIONS, 13,000 POUNDS GW, HOVER TO [illegible] [illegible] TO 90 KTS

	V_{KTS}	% TIME	CYCLES/ 1000 HRS	AFT CG
.33	30	.5	.1653 x [illegible]	13000
.30	45	.05	.0165 x [illegible]	15300
.28	60	.04	$.0132 \times 10^6$	15500
.20	75	.03	$.0099 \times 10^6$	12700
.14	90	.03	$.0099 \times 10^6$	10000

TABLE A2.8. HTR-XV-15 MISSION PROFILE (CONTINUED)

HTR-XV15 MISSION PROFILE (CONTINUED)

H. DECELERATIONS, 13,000 POUNDS GW, 35 KTS TO HOVER, 90 KTS TO HOVER

g's	V_{KTS}	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125 R AFT CG	FWD CG
-.34	30	.5	$.1653 \times 10^6$	9000	
-.39	90	.03	$.0099 \times 10^6$	47000	
-.44	75	.03	$.0099 \times 10^6$	36500	
-.36	60	.04	$.0132 \times 10^6$	22000	
-.34	40	.05	$.0165 \times 10^6$	11500	

I. SIDEWARD & REARWARD FLIGHT

g's	V_{KTS}	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
	20	1.65	$.1 \times 10^6$	---	25000
	45	1.65	$.091 \times 10^6$	---	60000

J. CLIMB IN HOVER, MAXIMUM LOAD AT FORWARD C.G.

GROSS WEIGHT	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
10500	.667	$.2204 \times 10^6$	---	47300
13000	.667	$.2204 \times 10^6$	---	45200
15500	.667	$.2204 \times 10^6$	---	41800

TABLE A2.8. HTR-XV-15 MISSION PROFILE (CONTINUED)

HTR-XV15 MISSION PROFILE (CONTINUED)

III CONVERSION - 13,000 POUNDS GW, 9 30% TOTAL TIME

A. LEVEL FLIGHT, FORWARD AND AFT C.G. [1]

V_{KTS}	$i_{N°}$	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125 R AFT CG	FWD CG
100	75	.9	$.2975 \times 10^6$	25000	28500
130	75	.65	$.2149 \times 10^6$	30000	36500
110	60	.75	$.2480 \times 10^6$	27500	28000
140	60	.5	$.1653 \times 10^6$	30000	33000
120	45	.75	$.2295 \times 10^6$	29000	30000
150	45	.5	$.1530 \times 10^6$	35000	34000
[illegible]	30	.75	$.2111 \times 10^6$	14500	14000
160	30	.5	$.1407 \times 10^6$	38000	44000
130	15	.75	$.1922 \times 10^6$	9000	19000
170	15	.5	$.1231 \times 10^6$	34000	38000

B. TURNS, 0→60° [2], FORWARD & AFT C.G.

V_{KTS}	$i_{N°}$	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
80	75	.15	$.0496 \times 10^6$	27800	56200
120	75	076	$.0251 \times 10^6$	[illegible]	60400
90	60	.166	$.0549 \times 10^6$	33000	63000
110	60	.084	$.0278 \times 10^6$	35500	49000
110	45	.166	$.0508 \times 10^6$	32200	43500
130	45	.084	[illegible]	39700	45700
130	30	.166	$.0467 \times 10^6$	27500	35000
150	30	.084	$.0236 \times 10^6$	32500	35000
130	15	.166	$.0425 \times 10^6$	26000	24500
160	15	.084	$.0215 \times 10^6$	67200	62800

C. CLIMBS, 0→4500 FT/MIN [2], AFT CG

V_{KTS}	$i_{N°}$	% TIME	CYCLES/ 1000 HRS	AFT CG
80	75	.15	$.0496 \times 10^6$	15600
100	65	.2	$.0661 \times 10^6$	18500
120	45	.2	$.0612 \times 10^6$	30000
130	30	.2	$.0563 \times 10^6$	16800
140	10	.275	$.0680 \times 10^6$	18400

D. DESCENTS, AFT CG

RATE [2]	V_{KTS}	$i_{N°}$	% TIME	CYCLES/ 1000 HRS	AFT CG
0→2000 FT/MIN	80	75	.08	$.0264 \times 10^6$	34500
0→3500 FT/MIN	100	65	.1	$.0331 \times 10^6$	41500
0→5500 FT/MIN	120	45	.1	$.0306 \times 10^6$	36500
0→5000 FT/MIN	130	30	.1	$.0281 \times 10^6$	22000
0→3000 FT/MIN	140	10	.12	$.0297 \times 10^6$	21000

TABLE A2.8. HTR XV-15 MISSION PROFILE (CONTINUED)

HTR-XV15 MISSION PROFILE (CONTINUED)

IV. AIRPLANE FLIGHT - 13,000 POUNDS GW, 43% TOTAL TIME

A. LEVEL FLIGHT, FORWARD AND AFT C.G.[1]

δ_F°	V_{KTS}	ALT.	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125 R AFT CG	± RESULTANT MOMENT AT .125 R FWD CG
40	140	SL	1.7	$.3937 \times 10^6$	8300	10000
40	180	SL	1.7	$.3937 \times 10^6$	26500	24500
20	180	SL	1.7	$.3937 \times 10^6$	7800	17500
0	180	SL	1.7	$.3937 \times 10^6$	10500	9700
0	200	SL	3.4	$.7874 \times 10^6$	14000	15000
0	240	SL	3.4	$.7874 \times 10^6$	12000	26500
0	260	SL	3.4	$.7874 \times 10^6$	10000	30000
0	180	5000'	2.55	$.5906 \times 10^6$	1000	10700
0	200	5000'	2.55	$.5906 \times 10^6$	15500	12000
0	240	5000'	2.55	$.5906 \times 10^6$	15000	22000
0	260	5000'	2.55	$.5906 \times 10^6$	11800	26800
0	200	10000'	1.36	$.3150 \times 10^6$	20000	12500
0	240	10000'	1.36	$.3150 \times 10^6$	16500	20000
0	260	10000'	1.36	$.3150 \times 10^6$	13300	24500
0	280	10000'	1.36	$.3150 \times 10^6$	11300	27000
0	300	10000'	1.36	$.3150 \times 10^6$	10000	30000

B. TURNS, 0 60°[1,3], FORWARD AND AFT C.G.

δ_F°	V_{KTS}	ALT.	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
0	140	5000'	.7	$.1621 \times 10^6$	24400	26200
0	220	5000'	.36	$.0834 \times 10^6$	18100	19300
0	280	5000'	.12	$.0278 \times 10^6$	21800	31600
0	140	10000'	.7	$.1621 \times 10^6$	29000	26000
0	220	10000'	.36	$.0834 \times 10^6$	20900	22200
0	280	10000'	.12	$.0278 \times 10^6$	14700	29300

C. PULLUPS

g's	V_{KTS}	ALT.	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
2.26	210	5000'	.0225	$.0052 \times 10^6$	26000	
2.18	210	10000'	.0225	$.0052 \times 10^6$	28500	
1.48	270	5000'	.0075	$.0017 \times 10^6$	13000	
1.68	270	10000'	.0075	$.0017 \times 10^6$	15000	

D. PUSHOVERS

g's	V_{KTS}	ALT.	% TIME	CYCLES/ 1000 HRS	AFT CG	FWD CG
-.5	210	5000'	.0225	$.0052 \times 10^6$	33500	
-.5	210	10000'	.0225	$.0052 \times 10^6$	30000	
-.5	270	5000'	.015	$.0035 \times 10^6$	27500	
-.5	270	10000'	.015	$.0035 \times 10^6$	22000	

TABLE A2.8. HTR-XV-15 MISSION PROFILE (CONTINUED)

C-5

HTR-XV15 MISSION PROFILE (CONTINUED)

	g's	V_{KTS}	ALT.	% TIME	CYCLES/ 1000 HRS	± RESULTANT MOMENT AT .125 R AFT CG	FWD CG
E. ACCELERATIONS							
	.36	95	0	.1	$.0232 \times 10^6$	25400	
	.35	95	5000'	.2	$.0463 \times 10^6$	25500	
	.27	120	0	.2	$.0463 \times 10^6$	11500	
	.29	120	5000'	.2	$.0463 \times 10^6$	13000	
	.29	120	10000'	.2	$.0463 \times 10^6$	15000	
	.20	140	0	.2	$.0463 \times 10^6$	8300	
	.23	140	5000'	.2	$.0463 \times 10^6$	11500	
	.24	140	10000'	.2	$.0463 \times 10^6$	14000	
F. ACCELERATIONS							
	-.4	120	0	.15	$.0347 \times 10^6$	14800	
	-.4	120	5000'	.3	$.0695 \times 10^6$	19100	
	-.4	120	10000'	.3	$.0695 \times 10^6$	22000	
	-.4	140	0	.15	$.0347 \times 10^6$	8000	
	-.4	140	5000'	.3	$.0695 \times 10^6$	14500	
	-.4	140	10000'	.3	$.0695 \times 10^6$	19500	

1 1/2 TIME AT FORWARD C.G. AND 1/2 TIME AT AFT C.G.

2 MAXIMUM ALTITUDE LOADS SELECTED OVER RANGE OF BANK ANGLES OR RATES OF CLIMB/DESCENT. POWER LIMIT AND WING STALL ARE BOUNDARIES.

3 MAXIMUM ALTITUDE LOADS SELECTED OVER RANGE OF BANK ANGLES. WING STALL IS A BOUNDARY.

TABLE A2.8. HTR-XV-15 MISSION PROFILE (CONTINUED)

HTR XV-15: LIFE CALCULATIONS FOR DESIGN 37 BLADE & HUB

1. DAMAGE IS CALCULATED BASED ON LOADS FROM THE MISSION PROFILE AT .125 R.

2. THE DESIGN E.L. AT .125 R ARE ± 64,800 IN.-LBS. RESULTANT MOMENT FOR THE HUB AND ± 46,500 IN.-LBS RESULTANT MOMENT FOR THE ROOT END.

ROOT END

CONDITION	APPLIED LOAD (± IN.-LBS)	LOAD E.L.	n CYC./1000 HRS	N 1 (CYCLES)	n/N DAMAGE/ 1000 HRS
HOVER					
7° CYCLIC MANEUVER	59400	1.277	26448	6 x 10	.004408
5.6° CYCLIC MANEUVER	48500	1.043	129596	6.6 x 10	.001964
80 KT TURN					
45°	49500	1.065	33100	5.2 x 10	.000637
60°	63000	1.355	16500	2.9 x 10	.005690
80 KT PULLUP	49400	1.062	72732	5.2 x 10	.001399
DECEL AT 90 KT	47000	1.011	9918	8.6 x 10	.000115
SIDEWARD FLT	60000	1.290	91000	5.8 x 10	.015690
CLIMB	47300	1.017	220400	8 x 10	.002755
TRANSITION					
75° i_N TURN					
80K	56300	1.211	24795	1.1 x 10	.002254
120K	60400	1.299	12563	5 x 10	.002513
60° i_N TURN					
90K	63000	1.355	27440	2.9 x 10	.009462
110K	49000	1.054	13885	5.2 x 10	.000267
15° i_N TURN					
160K	67200	1.445	10760	1.5 x 10	.007173
160K	62800	1.351	10760	3.2 x 10	.003363
					Σ.057690

$$\text{BLADE LIFE} = \frac{1000\ \text{HRS}}{.057690} = 17{,}334\ \text{HOURS}$$

HUB

CONDITION	APPLIED LOAD (± IN.-LBS)	LOAD E.L.	n CYC./1000 HRS	N 1 (CYCLES)	n/N DAMAGE/ 1000 HRS
TRANSITION 15° i_N TURN, 160 KT	67200	1.037	10760	1.2 x 10	.008967

$$\text{HUB LIFE} = \frac{1{,}000\ \text{HRS}}{.008967} = 111{,}524\ \text{HRS}$$

TABLE A2.9. LIFE CALCULATIONS FOR DESIGN 37 BLADE & HUB

APPENDIX III

XV-15 FLY-BY-WIRE PRELIMINARY DEVELOPMENT SPECIFICATION

The following is the Preliminary Development Specification used to solicit vendor technical and budgetary cost responses for the development of the fly-by-wire flight control system for the XV-15. It was released to vendors as Boeing Vertol Document D210-11256-1.

The Flight Control System comprises the Primary Flight Controls System (PFCS) and the Stability and Control Augmentation System (SCAS).

For the purpose of definition for this document, the primary flight control system is considered to include the control transducers, electronics, rotor actuators, airplane surface actuators, and control panels. The SCAS is considered to include the SCAS sensors and the SCAS electronics. This document contains or references all necessary specifications and/or characteristics germane to the flight control system design intended to be used in modifying the XV-15 aircraft to a hingeless rotor configuration.

Since a tilt-rotor aircraft will operate as a fixed-wing aircraft, rotary-wing aircraft, as well as combinations thereof, it does not conform entirely to any of the types of aircraft defined in current specifications. Therefore, the basic design criteria presented herein will reflect considerations unique to tilt prop/rotor operations. Reference is made to FAR XX and applicable equipment specifications.

FLIGHT CONTROL SYSTEM REQUIREMENTS

1.0 SYSTEM DEFINITION

The PFCS shall provide for direct pilot control of the tilt-rotor aircraft by control of rotor blade pitch via swashplates, airplane surfaces, and engine performance. The system shall modify pilot control inputs as a function of nacelle incidence angle and rotor speed. The PFCS shall accept inputs from the SCAS for aircraft stability and maneuver enhancement. The SCAS shall provide rate and attitude stabilization in pitch, roll and yaw, and provide gust alleviation signals to PFCS.

2.0 PFCS DESCRIPTION

Pilot input shall be via conventional dual mechanically synchronized controls comprising a longitudinal/lateral control stick, directional pedals, and an engine throttle control. Signals proportional to control position shall be generated by linear or rotary transducers connected to each control. The position signals shall be process in the control unit to generate commands for the rotor control actuators (3 per rotor), the flaperon actuators (one each side), the rudder actuator, elevator actuator, and engine N_1 control actuators (Figure A3.1).

2.1 SYSTEM FUNCTIONS

Major system functions shall be as shown in Figure A3.2 and described in the following paragraphs.

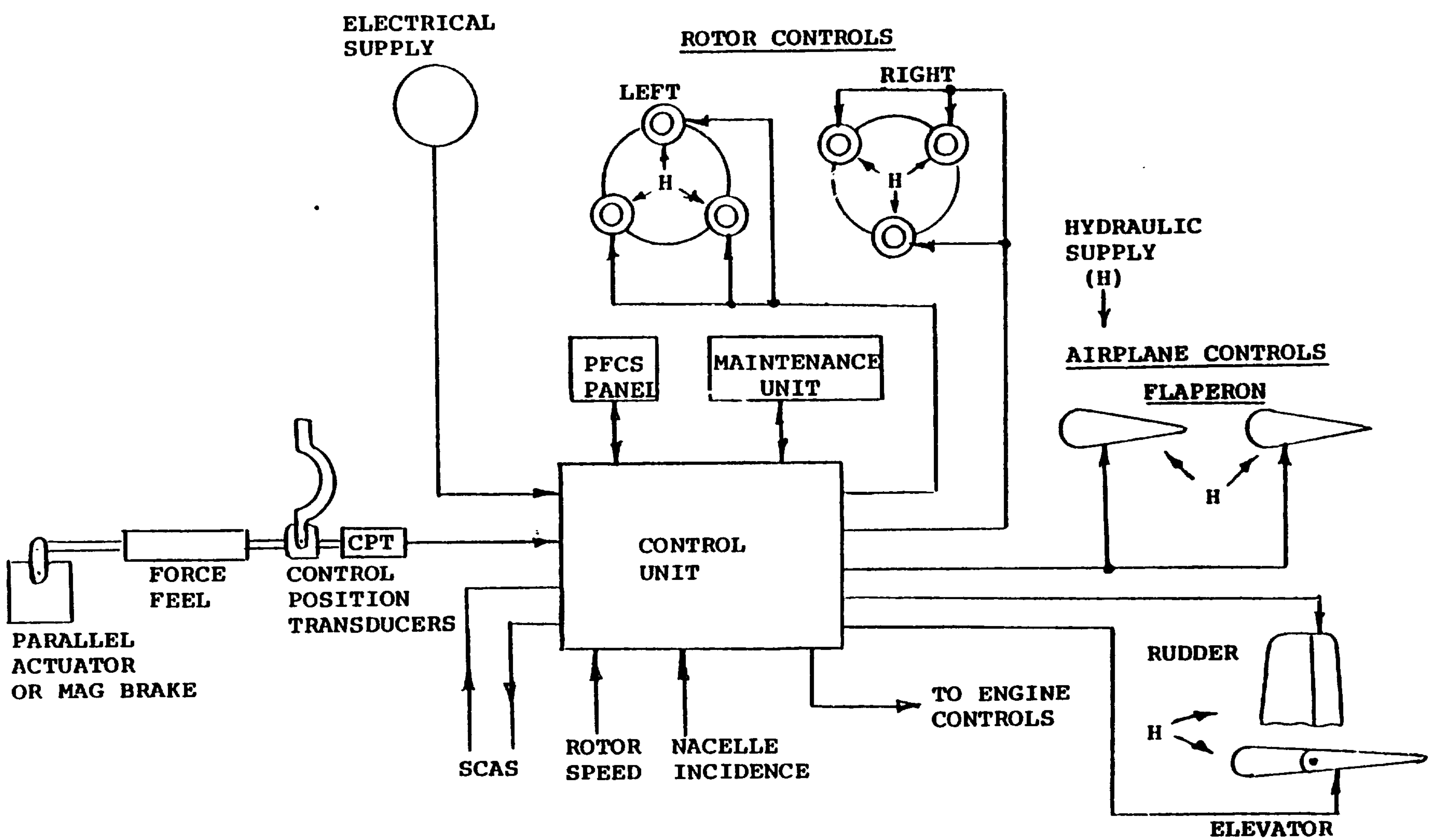


FIGURE A3.1. FLY-BY-WIRE PRIMARY FLIGHT CONTROL SYSTEM

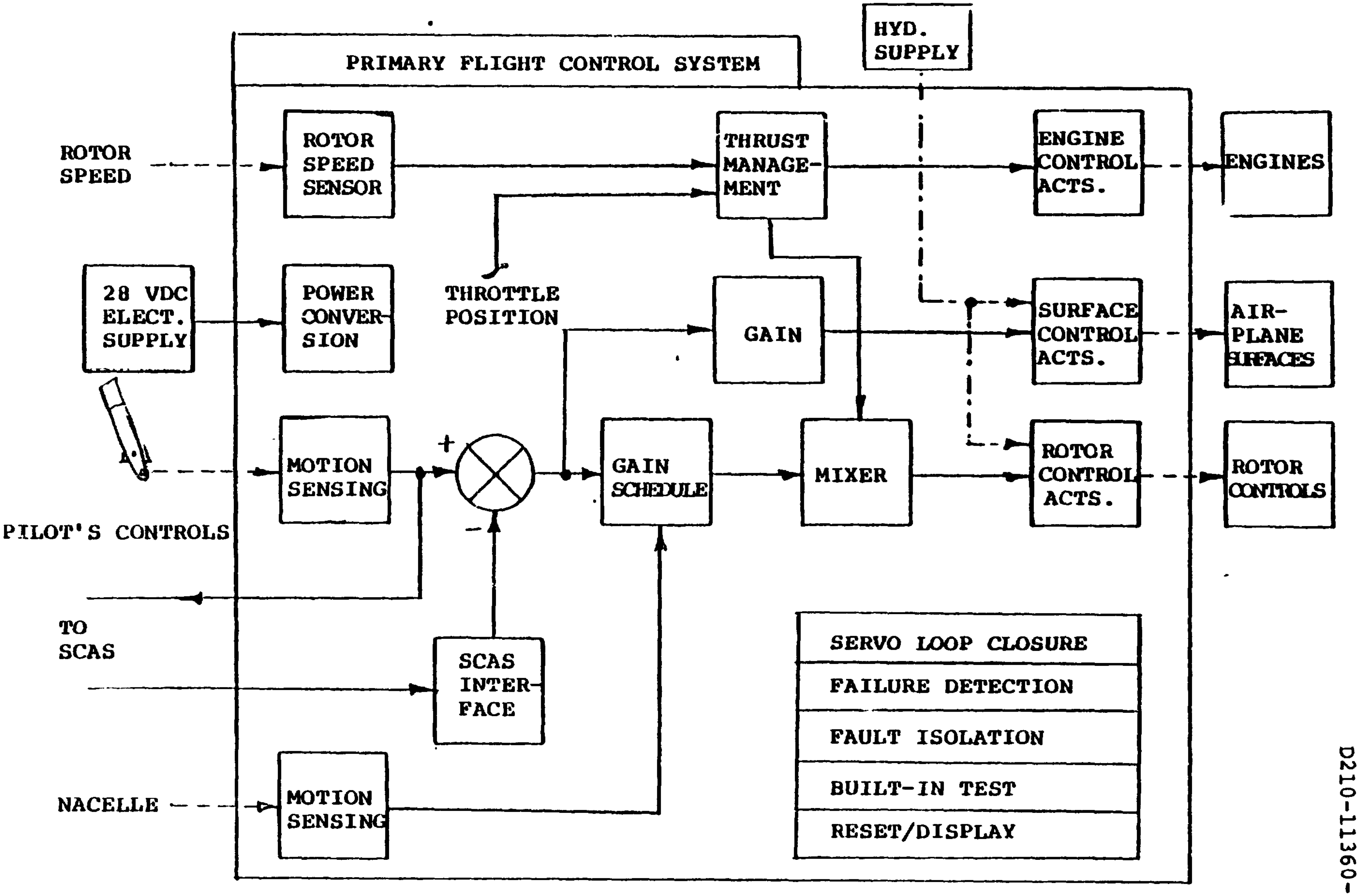


FIGURE A3.2. PRIMARY FLIGHT CONTROL SYSTEM FUNCTIONS/INTERFACE

a. <u>Motion Sensing</u>. The control position transducers shall convert pilot stick and pedal motions to equivalent electrical signals for input to the demodulator circuitry.

b. <u>Signal Conditioning</u>. The control unit shall convert the transducer signals to the appropriate form for transfer to the mixing curcuitry and to the SCAS.

c. <u>SCAS Interface</u>. The control unit shall accept SCAS commands via authority and rate limit networks. The limited signals shall be summed with the demodulated control position signals before mixing.

d. <u>Gain Scheduling</u>. Axis command signals (summation of pilot contol and SCAS command) shall be scheduled as a function of nacelle angle. In general, pilot inputs to the rotor are phased out as the nacelle is brought to the horizontal position (zero degrees).

e. <u>Thrust Management</u>. Shall control engine performance through engine N_1 controls and rotor collective pitch in response to pilot throttle setting rotor rpm and manual trim inputs to vary rpm and differential collective pitch. Direct pilot control of collective pitch is phased out at zero degrees nacelle incidence.

f. <u>Airplane Surface Control</u>. Axis commands shall be processed via appropriate gains and actuation to position the flaperons, rudder, and elevator.

g. Mixing. The control unit shall mix scheduled axis commands and governor outputs via appropriate gains to position the rotor control actuators.

h. Servo Loop Closure. The control unit shall include the electronics to control rotor, airplane surface, and engine control actuators.

i. Rotor Actuation. The rotor control actuator shall convert the mixer outputs to equivalent rotor swashplate motion.

j. Power Conversion. The control unit shall convert the 28VDC supply to AC for sensor excitation and DC supplies as needed to operate electronic devices used in the system.

k. Failure Detection. Each control unit shall process all failure detection within its channel and, upon detecting a failure, shut down the channel inputs to the affected actuators, and transmit failure information to the PFCS/ maintenance panels.

l. PFCS Panel/Maintenance Unit. Shall provide pilot input, monitoring, display, and test capability to:

- Set rotor rpm variation relative to fixed schedule.
- Adjust rotor torque balance by manual input to differential collective pitch.
- Determine the operable paths within the system.
- Provide logic to drive aircraft caution/advisory panel.
- Reset failed channels within the system (if failure has cleared).
- Conduct GO/NO GO ground tests on each channel of the system.

m. Fault Isolation. The maintenance unit shall provide readouts to indicate location of system failure to assist in isolation of faults to a line replaceable unit.

2.2 MAJOR COMPONENT RESPONSIBILITIES

The following is a regrouping of PFCS functions by major component.

a. Control Position Transducers
- Motion sensing

b. Control Unit
- Signal conditioning and buffering
- SCAS interface
- Gain scheduling
- Thrust Management
- Mixing
- Servo loop closure
- Power conversion
- failure detection

c. Rotor Control Actuator
- Rotor swashplate actuation

d. Airplane Surface Actuators
- Flaperon actuation
- Rudder actuation
- Elevator actuation

e. PFCS Panel

- Fault reset
- Fault display
- Manual rpm control
- Manual torque matching (differential collective pitch)

f. Maintenance Unit

- Fault isolation display
- Built-in test control

2.3 EXTERNAL INTERFACES

The PFCS shall be designed to interface with the following equipment and subsystems of the aircraft.

a. Pilot's Stick, Pedals, and Throttle Lever. This is a mechanical interface with the control position transducers. The existing mechanical XV-15 controls will be adapted to achieve this interface.

b. SCAS. This is an electrical interface with the control unit.

c. Nacelle Incidence. This is the mechanical interface with nacelle.

d. Rotor Speed. This is a mechanical interface with the rotor accessory gearbox.

e. Rotor Swashplate. This is the mechanical interface with the rotor system.

f. Airplane Surface. This is the mechanical interface with the flaperon, rudder, and elevator.

g. Engine. This is the mechanical interface of the linkage controlling N_1 control inputs.

h. Electrical Power Supplies. This is the electrical interface with the 28VDC electrical power supply.

i. Hydraulic Power Supply. This is the mechanical interface with the rotor and airplane surface control actuators.

2.4 REDUNDANCY MANAGEMENT

In order to meet the reliability goals specified therein, the PFCS shall be at least single-fail operative, which is defined to mean that the system shall withstand any one failure in the system.

Failure detection logic shall be dualized where necessary to meet reliability goals. Dual logic shall be used to drive dual-failure warning circuits.

Details on mechanization of the redundancy management shall be as specified in Paragraph 5.0, "Major Component Characteristics".

2.5 FAILURE DETECTION

The PFCS shall have a self-contained capability for ground checkout.

Each channel shall identify in-flight failures independently and furnish signals to the control logic and panels for appropriate system corrective action and crew notification.

Details of failure detection shall be as specified in Paragraph 5.0, "Major Component Characteristics and Requirements".

3.0 SYSTEM CHARACTERISTICS

3.1 SYSTEM PERFORMANCE

3.1.1 Gains, Schedules, Transfer Functions. Shall be as defined in Section 5.0 of this appendix.

3.1.2 Accuracy. The system electronics supplier shall be responsible for analysis and control of system tolerances so that the overall system (pilot control to control actuator) accuracy tolerances are maintained. To this end, the system electronic supplier will support definition of control position transducer (CPT) and actuator performance.

a. Static Gain Accuracy. The average gain for all control units shall be within 2% of the values specified in Section 5.0, the rotor speed control loops shall be within .75% of the value specified. The static gain of individual control units shall be within 1.5% of the average. For a given control input, the accuracy is defined as the percentage difference between the desired actuator position and the actual actuator position. These accuracies include schedule accuracies.

b. System Null. The total steady state null associated with the PFCS (sensor to actuator) shall not exceed .020 in actuator.

c. Resolution. Resolution is defined as the minimum change in control required to obtain actuator motion. The resolution (equated in actuator motion) shall not exceed .002 inch rotor actuator, or airplane surface actuator.

d. System Hysteresis. Hysteresis within the PFCS shall not exceed .004 inch actuator for rotor control or airplane surface control paths.

e. Cross Coupling. Full motion of any axis or combination of axes shall not require more than two percent of full control displacement (in axes not in motion) to compensate.

3.1.3 Actuator Frequency Response

a. Rotor Control Actuators. The rotor control actuator shall exhibit a second order response with a natural frequency of 40 rad/sec and damping factor of .7. This response shall be achieved while driving a rotor load represented as a second order response with a natural frequency of 35 rad/sec and damping factor of .18. This response shall be achieved with a tensile or compressive load of 0 to 1700 pounds while not exceeding velocity limit of 2.5 in./sec. Additional deviations from linear performance will be defined later.

b. Airplane Surface Actuators. (To be defined in follow-on phase).

c. Engine Control Actuators. (To be defined in follow-on phase).

3.1.4 Failure Detection and Effects. The PFCS shall include the following requirements relating to system failures and effects.

a. Failure Tolerant Performance. The PFCS shall be designed so that the aircraft meets the failure tolerance performance of FAR XX .671, Subparagraph (c).

b. Failure Detection and Isolation. Operation of the redundant channels shall be monitored to detect any failure or malfunction that could cause unsafe flight or system degradation requiring maintenance action. Unsafe flight is referred as loss of control or degradation of control (transient or steady state) that jeopardizes the pilot's ability to abort and land safely.

After the detection of failure, the failed channel shall be automatically inhibited from affecting the correctly operating channel(s). The detection and isolation time shall be compatible with Paragraph 3.1.3.c.

c. Failure Transients. Transients following first and second failures within the PFCS shall not exceed the limits shown in Table A3.1. Failure transients shall be defined by Figure A3.3. Limits defined do not take into account corrective action supplied by SCAS.

d. Failure Detection Threshold. The failure detection threshold must be set low enough to detect passive failures with normal system disturbances, detect valid failures, and minimize failure transients. The threshold must be high enough to minimize nuisance trips due to normal channel tolerances and transients.

e. Redundancy of Monitoring and Correction Circuitry. The detection, logic, and switching circuitry reliability shall be included in the channel reliability requirements. The failure or malfunction of the logic and switching circuitry shall be interpreted as a channel failure.

3.2 SYSTEM PHYSICAL CHARACTERISTICS

3.2.1 Control Device and System Loads

a. Control System Loads. The PFCS shall be designed to meet applicable portions of FAR XX .395 (considering there is no longer a linkage which can carry loads between the pilot's

FAILURE TYPE	ALLOWABLE SS OFFSET IN (a)	TIME DELAY t_o SEC	ALLOWABLE TRANSIENT-MAX $\int_{t_o}^{t_a} (\lvert\delta\rvert - \lvert a\rvert)\, dt$
Longitudinal • Rotor • Elevator	TBD	TBD	TBD
Lateral • Rotor • Flaperon	↓	↓	↓
Directional • Rotor • Rudder	↓	↓	↓
Throttle	↓	↓	↓
Rotor Actuator	↓	↓	↓
Flaperon Actuator	↓	↓	↓

a = maximum allowable steady state actuator position offset from commanded position

δ = actuator travel due to failure at time t

t_o = time delay

t_a = time when position offset from commanded input has been reduced to the maximum allowable steady state position offset

TABLE A3.1. FAILURE TRANSIENT LIMITS

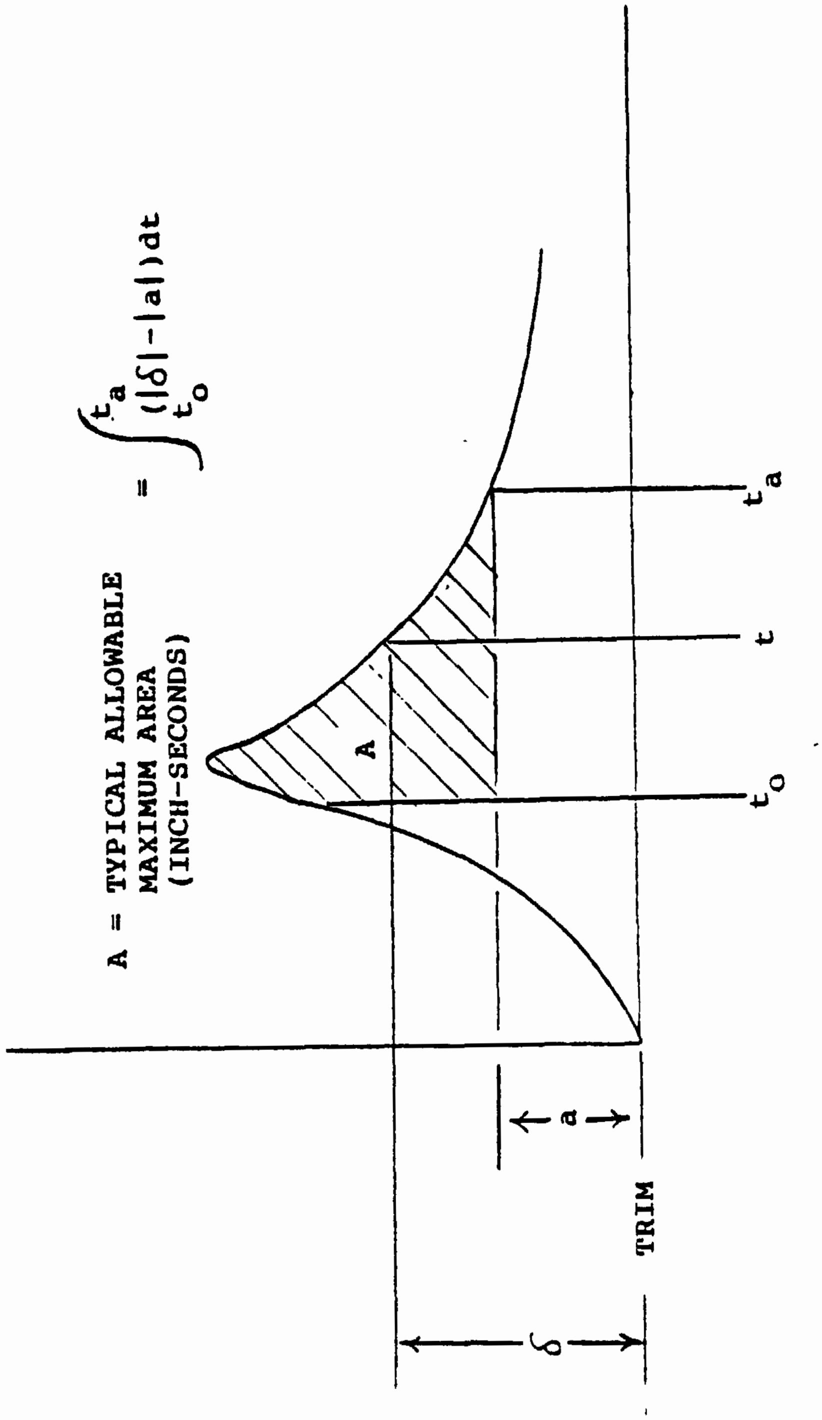


FIGURE A3.3. FAILURE TRANSIENT EFFECTS

control and the output actuator).

b. Limit Pilot Forces. The PFCS cockpit controls shall be designed to withstand the loads defined in FAR XX .397. The existing XV-15 controls will be adapted.

c. Dual Control System. The PFCS cockpit controls shall be designed to meet requirements of FAR XX .415.

3.2.2 System Packaging. PFCS components shall be packaged so that each channel is separately contained. System panels are an exception to this requirement.

Actuator sections shall be separated with respect to hydraulic supply.

System electronic assemblies shall be designed to facilitate changes during the development program. High density "production" packaging is not desired.

System component weights shall not exceed values defined below.

COMPONENT	TOTAL WEIGHT PER AIRCRAFT (LBS)
Control Unit (with Mounting Base)	100
Rotor Control Actuator	150
Airplane Surface Actuator	100
PFCS Panel	1
Maintenance Panel	4
Engine Control Actuator N_1	12

3.3 RELIABILITY

The primary flight control system, as defined in Figure A3.1, including the path from transducer input to actuator output and power supplies, but excluding cockpit mechanical controls, shall exhibit a flight safety reliability of .9999999 for a two-hour mission. Flight safety reliability is defined as the probability that the system will maintain the transfer functions defined for the system. In general, loss of flight safety will result in loss of the aircraft.

3.4 ENVIRONMENTAL CONDITIONS

3.4.1 Standard Condition. The following conditions shall be used to establish normal performance characteristics under standard conditions for making laboratory bench tests.

a. Temperature - room ambient 25 ±5°C (77°F ±9°F).

b. Altitude - normal ground.

c. Humidity - room ambient up to 90% relative humidity.

3.4.2 Environmental Service. Components of the PFCS shall meet the requirements of this specification under the conditions listed in the following paragraphs. Electronic components shall be tested under the conditions defined in MIL-E-5400 for Class 1A equipment. Actuators shall be tested to the conditions specified. The equipment supplier shall submit a detailed procedure to be approved by Boeing.

a. Altitude. Operation without degradation of performance throughout a pressure altitude range of -200 to +30,000 feet ASL per MIL-STD-810B.

b. Ambient Temperature. Operation throughout an ambient temperature range of -65° to +160°F.

c. Temperature Shock. Sudden changes in temperature of the surrounding atmosphere per MIL-STD-810B.

d. Humidity. Operation in a warm, highly humid atmosphere such as encountered in tropical areas per MIL-STD-810B.

e. Salt Fog. Operation is an atmosphere containing salt laden moisture per MIL-STD-810B.

f. Rain. Operation in a rain environment per MIL-STD-810B.

g. Sand and Dust. Operation in a dust (fine sand) laden atmosphere per MIL-STD-810B.

h. Immersion (for hydraulic actuators only). Operation after immersion in hydraulic fluid at a temperature of +275°F per MIL-C-5503.

i. Vibration. Operation during exposure to dynamic vibration stresses represented by those tests of MIL-STD-810B, Method 514.1, Procedure I, Part I, Equipment Category (A) to include:

- Resonance search
- Resonance dwell
- Cycling

j. Mechanical Shock. Operation after exposure to a mechanical shock environment similar to that expected in handling, transportation, and service use per MIL-STD-810B.

k. Electromagnetic Interference. Meeting per MIL-STD-461A.

4.0 DESIGN AND CONSTRUCTION

Electrical equipment shall conform with all applicable requirements of MIL-E-5400 for design,construction and workmanship except as modified herein. Hydromechanical equipment shall conform to the applicable requirements of MIL-H-5440, MIL-C-5503, and MIL-H-8775.

5.0 MAJOR COMPONENT CHARACTERISTICS AND REQUIREMENTS

5.1 DIRECT ELECTRICAL LINKAGE

5.1.1 Subsystem Description. The Direct Electrical Linkage (DEL) comprises the following units:

DEL Control Unit	-	Number of identical units per aircraft is the same as the channel redundancy level
Control Panel	-	1 per aircraft
Maintenance Unit	-	1 per aircraft
Control Position Transducers	-	4 per channel
Electrical Interconnecting Cables	-	To be supplied by Boeing Vertol

The DEL replaces not only the mechanical control linkages of the XV-15 aircraft, but also the five SCAS actuators, the two exciter actuators, the differential cyclic washout actuator, and the differential collective trim actuator.

5.1.1.1 DEL Control Unit. The DEL control unit synthesizes the following major flight control functions. The DEL control unit translates the cockpit control motions and the Stability and Control Augmentation (SCAS) signals into the appropriate actuator control signal inputs to provide manual and automatic flight control. Figures A3.4, A3.5, A3.6 and A3.7 provide the functional diagrams of the required control dynamics. Letters in circles adjacent to protions of the diagrams cross reference to the schedule callouts in Report CR-151950, Appendix F.

5.1.1.1.1 Transducer Signal Conditioning. The input signals from the control position transducers (CPTs), the nacelle incidence transducers, and the rotor speed transducers shall be buffered and suitably processed to the form and scaling required by the ensuing flight control computations.

5.1.1.1.2 SCAS Interface. The inputs from the dual SCAS shall be summed with the control position signals after processing through authority and rate limiting network and isolation buffer circuits designed to prevent propagation of any SCAS failure to the DEL. The inputs shall include rate/authority limit networks to limit responses to SCAS inputs.

5.1.1.1.3 Gain Scheduling. Axis command signals from the pilot controls and SCAS shall be scheduled as a function of nacelle angle in accordance with the transfer functions shown on Figures A3.4 through A3.7.

5.1.1.1.4 Thrust Management System. Control of engine performance and rotors collective blade pitch shall be synthesized in accordance with the transfer functions shown on Figure A3.7.

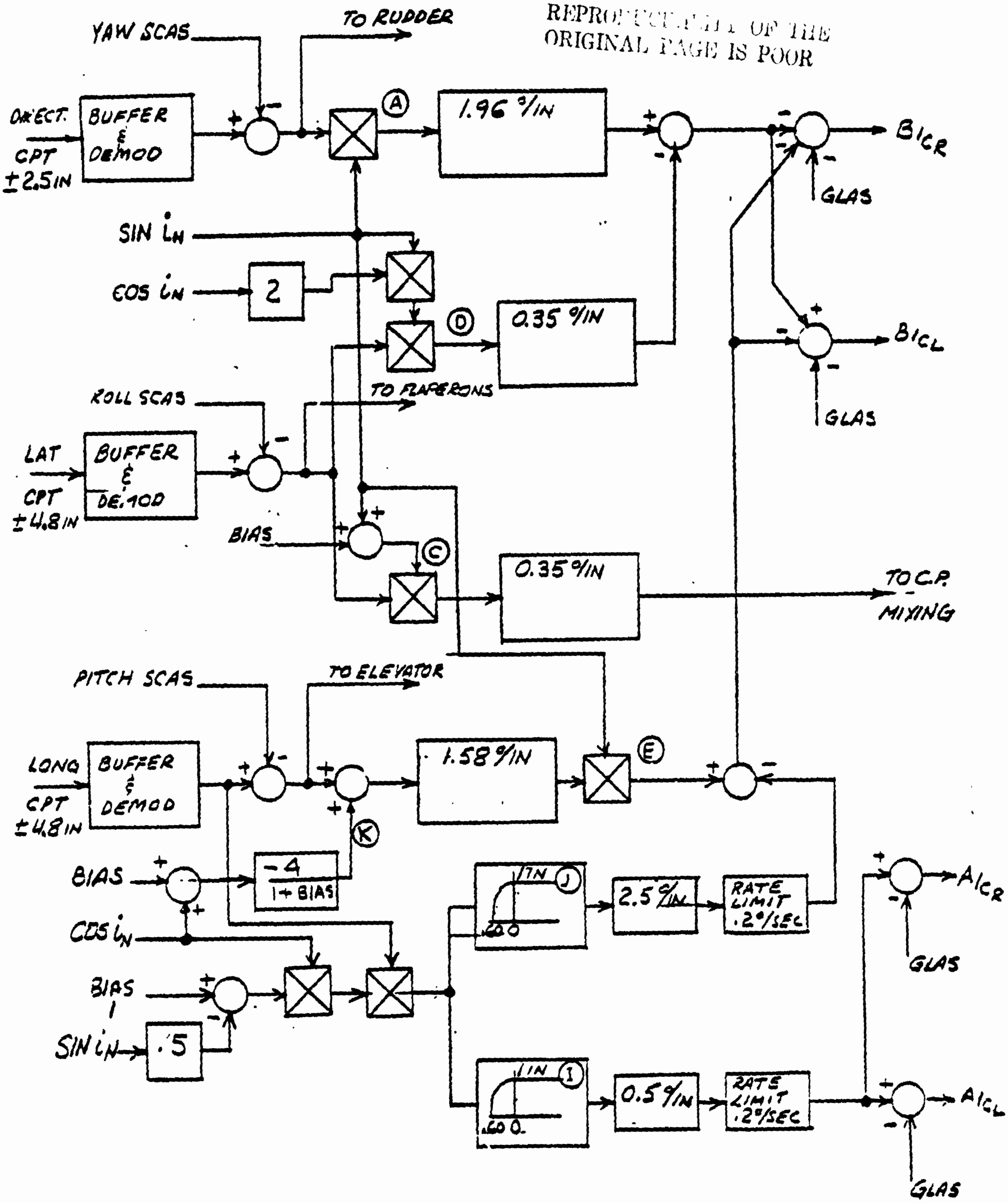


FIGURE A3.4. ROTOR CYCLIC CONTROLS

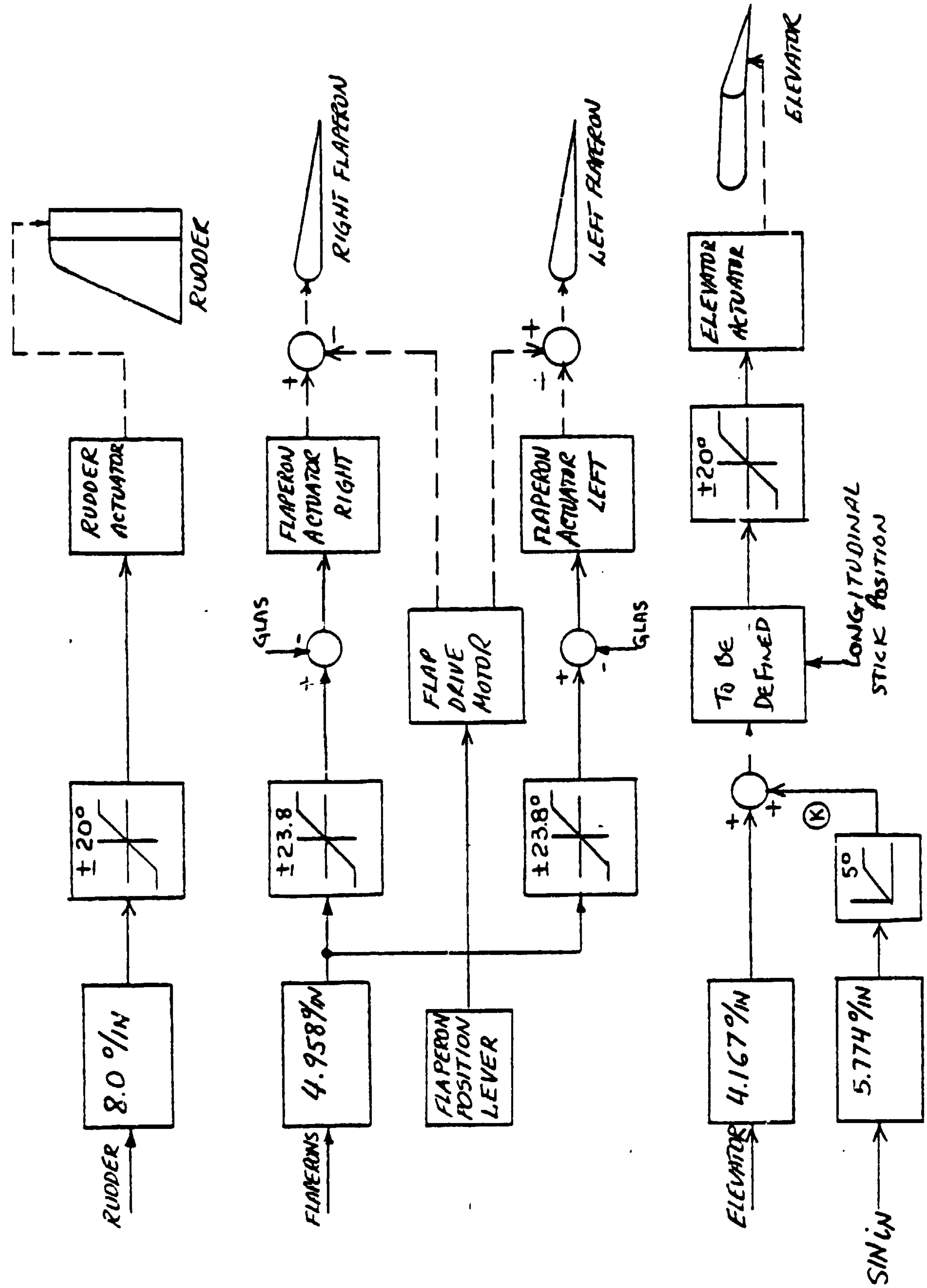


FIGURE A3.5. AIRPLANE SURFACE CONTROLS

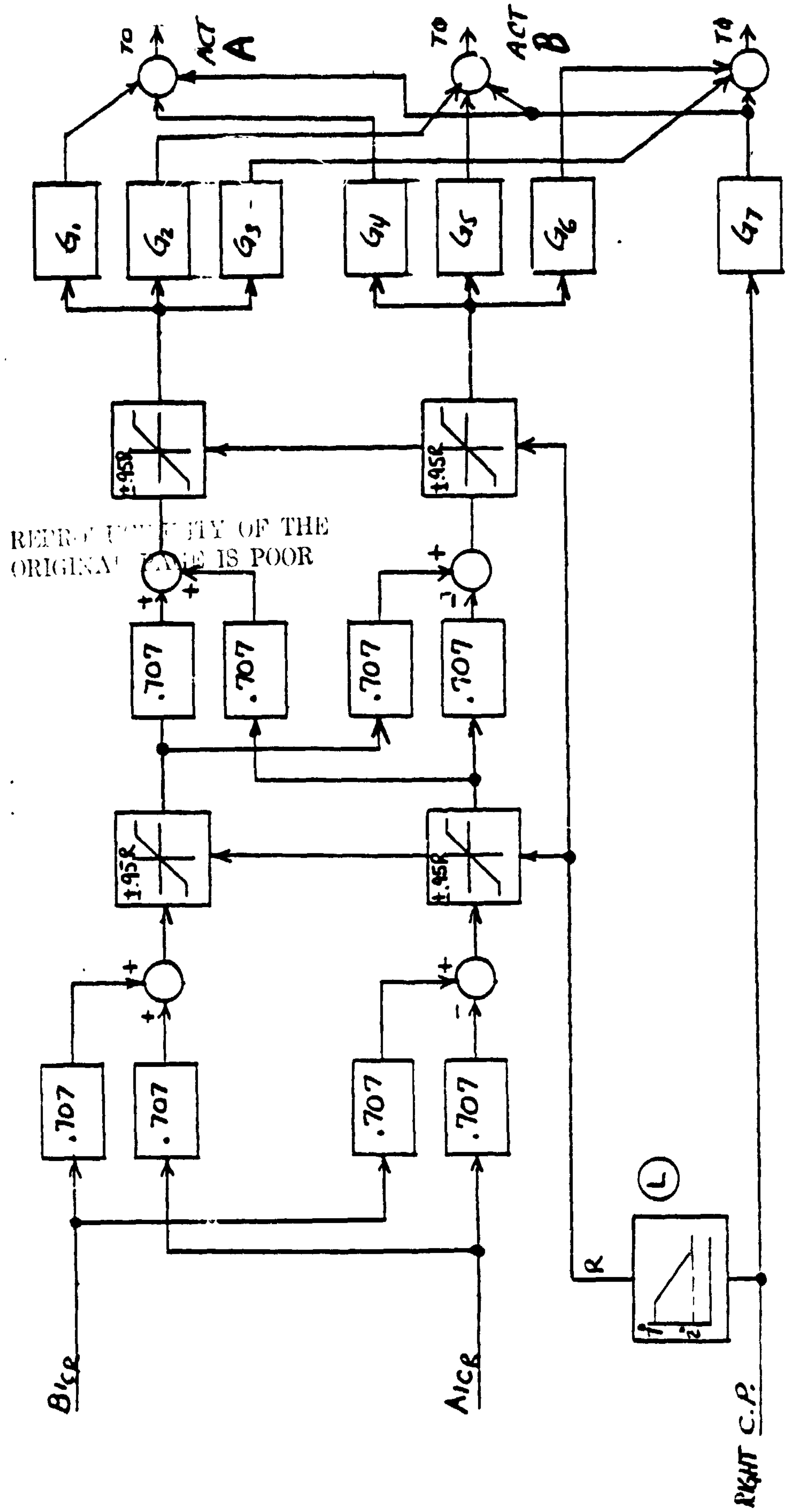


FIGURE A3.6. ROTOR CYCLIC CONTROL MIXING AND CUMULATIVE LIMITING (RIGHT ROTOR)

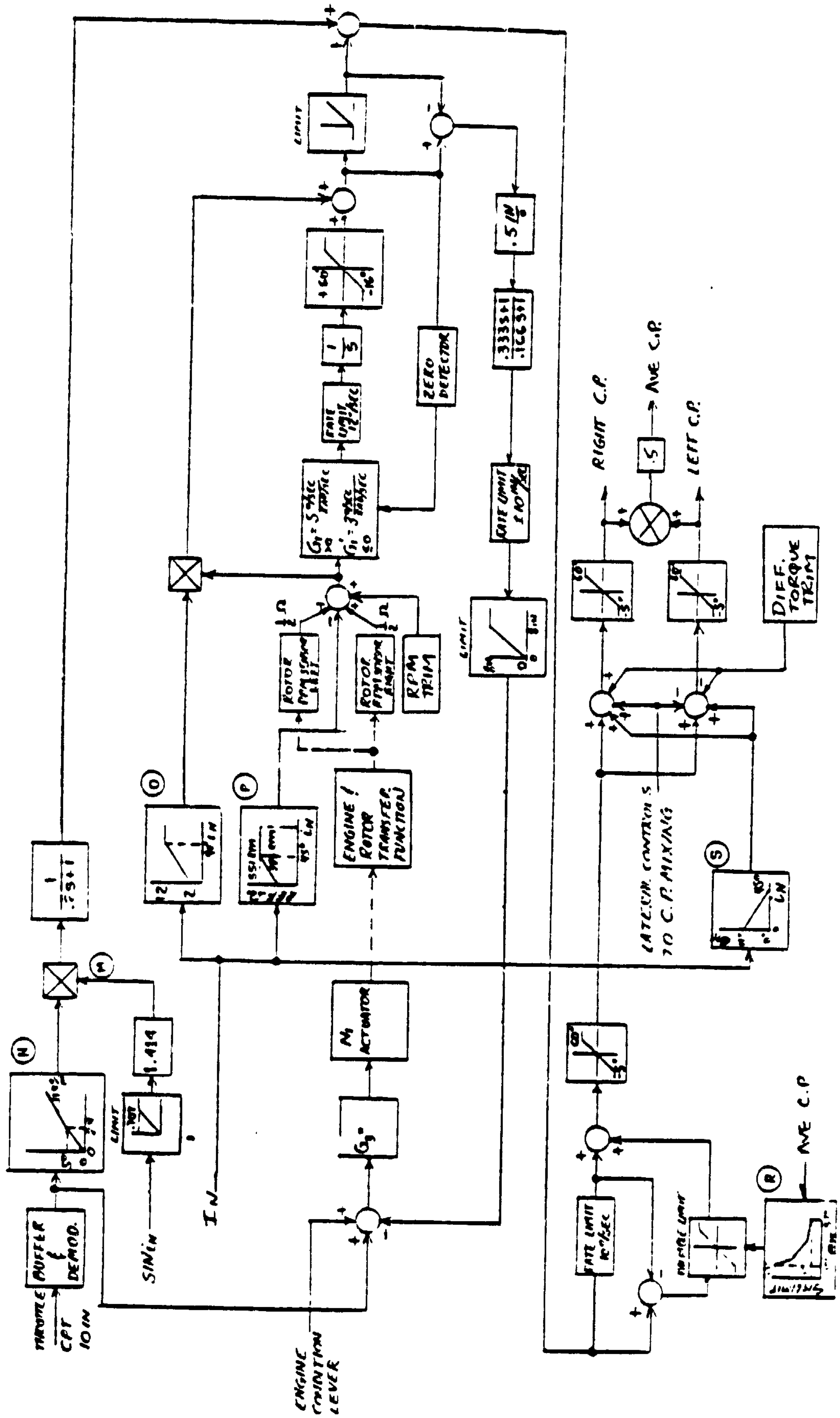


FIGURE A3.7. THRUST MANAGEMENT SYSTEM

5.1.1.1.5 Controls Mixing and Limiting. The scheduled axis commands and thrust management commands to the rotor actuators shall be mixed via appropriate gains and limited in accordance with the transfer diagram of Figure A3.6 to position the rotor actuators.

5.1.1.1.6 Airplane Surface Controls. The schedule airplane surface controls shall be processed via appropriate gains in accordance with the transfer diagrams of Figure A3.5 to position the actuators controlling the flaperons, rudder, and elevator.

5.1.1.1.7 Servo Loop Closure. The control unit shall include the actuator servo loop closure electronics to control the rotor, airplane surface, and engine control actuators.

5.1.1.1.8 Power Conversion. The externally supplied power to the control unit shall be 28 VDC. All other voltages required to power the Direct Electrical Linkage shall be generated within the control unit.

5.1.1.1.9 Failure Detection. Each control unit shall process all failure detection within its channel and upon detecting a failure, shut down the channel inputs to the affected actuators and transmit failure information to the PFCS/maintenance panels.

5.1.1.2 Control Panel. The control panel shall provide rpm trim, differential torque trim, fault annunciation and channel reset capability for the pilot.

5.1.1.3 Maintenance Unit. The maintenance unit shall provide the following functions in conjunction with the Built-In-Test Equipment (BITE).

- Determine the operable channels within the system.
- Provide logic to drive aircraft caution/advisory panel, and control panel.
- Conduct GO/NO GO ground tests on each channel of the system.
- Provide readouts to indicate location of system failure to assist in isolation of faults to a line replaceable unit.

5.1.1.4 Control Position Transducers. The control position transducers (CPTs) shall translate cockpit control motions into equivalent electrical signals which are in turn transmitted to the DEL control unit. The options can be exercised to consider either linear types or rotary types of transducers, also to consider either analog types such as LVDT or RVDT or digital types such as shaft encoders. The range or stroke of the transducers is dictated by their installation configuration.

5.1.1.5 Electrical Interconnecting Cables. The electrical interconnecting cable assemblies shall be flight control dedicated and prefabricated utilizing simple multi-conductor wires MIL-C-83723 self-locking threaded connectors, and appropriate strain relief. The system configuration shall be designed such as to use point-to-point cables to the extent possible. Figure A3.8 defines a tentative interconnect for a single channel.

5.1.1.6 Interfaces. The Direct Electrical Linkage shall interface with the following equipment and subsystems of the aircraft.

a. Cockpit Controls. This is a mechanical interface of control position transducers with the cockpit controls. The existing XV-15 cockpit controls will be retained. The cockpit controls are longitudinal/lateral stick, the direction pedals, and the throttle lever. Figure A3.1 shows the DEL interface with the cockpit controls.

b. Actuators. This is an electrical interface with the control power actuators. The interface shall consist of the actuator command error and the actuator position electrical feedback. The following control power actuators shall be interfaced with the DEL.

Rotor Swashplate Actuators - Electrohydraulic

3 left rotor

3 right rotor

Airplane Surface Actuators - Electrohydraulic

Rudder

Flaperon right

Flaperon left

Elevator

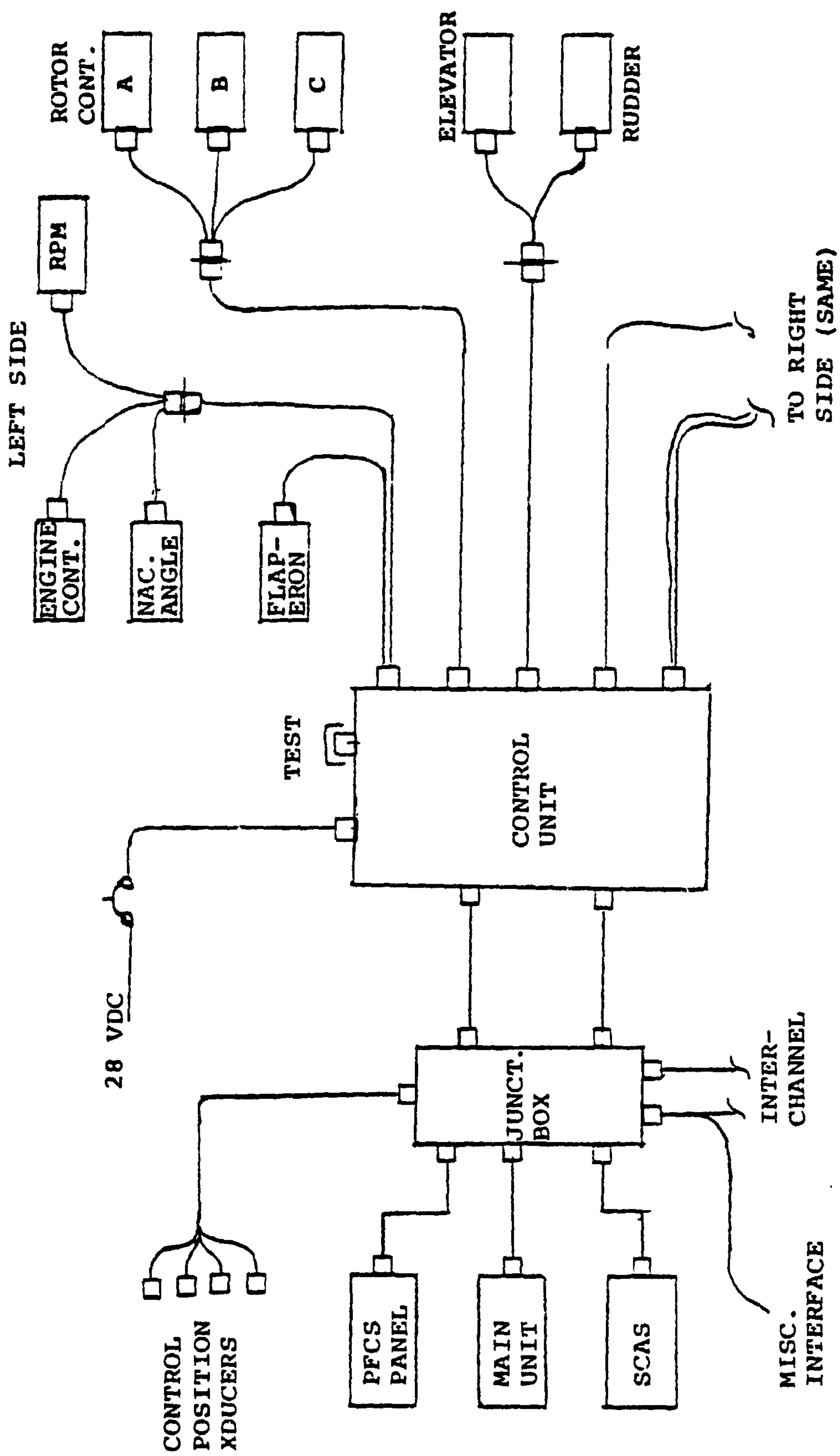


FIGURE A3.8. PRIMARY FLIGHT CONTROL SINGLE CHANNEL INTERCONNECT

Engine N_1 Actuator - Electromechanical

Right engine

Left engine

c. Electrical Power Supply. This is the interface with the 28 VDC supply. The supply may vary according to limits defined in MIL-STD-704A, Category B.

d. SCAS. This is the electrical signal interface with the stability and control augmentation system.

e. Sensors Interface. This is the electrical signal interface with the nacelle incidence sensors and the rotor speed sensors. The nacelle incidence sensor shall be a synchro providing a signal proportional to the sine and to the cosine of the nacelle incidence cycle. The rotor speed sensors shall be proximity switches in the transmission providing pulses the frequency of which is proportional to rotor speed.

5.1.2 System Performance

System gain, schedule, and transfer function accuracies shall be met over the range of environments defined in Paragraph 3.4. Components shall be designed so that the overall system meets the requirements defined in Section 6.3.1.

5.1.3 Redundancy

a. The Direct Electrical Link (DEL) shall be at least single fail operative for any failure. Electrical supply failures shall be considered as failures of the DEL.

b. All failures causing loss of one DEL channel shall be detected and immediately displayed.

c. A failed channel shall be automatically removed from the system as soon as necessary to maintain flight control operation.

d. Failure detection and warning/display logic shall be dualized where necessary, to meet reliability requirements.

e. The allowable transient due to a failure shall meet the requirements of Paragraph 3.1.4.

5.1.4 Reliability

The overall system safety reliability of the PFCS shall be as defined in Paragraph 3.3 for a two-hour flight. Reliability shall be demonstrated by analytical methods based on known failure rates of components used in the design. The required redundancy level shall be adopted to meet this reliability requirement.

5.1.5 Implementation Options

The following options shall be available for the design of the DEL electronics.

a. Analog sensors and signal processing, including control scheduling.

b. Digital sensors and signal processing.

c. A combination of analog and digital signal processing and sensors.

If the digital implementation option is adopted, a detail software development and control design must be included with the hardware design. Generic failures such as computer overflow must be covered in the design.

5.1.6 Diagnostics

5.1.6.1 Failure Detection and Display. The DEL shall contain the capability to detect and display any malfunction causing unsafe operation or degradation of operation occuring in the primary flight control system, including actuator failures.

5.1.6.2 Built-In-Test Equipment (BITE). The DEL shall have sufficient built-in test equipment to localize any failure to a line replaceable unit (LRU).

5.1.6.3 System Checkout. The maintenance unit, in conjunction with the BITE, shall provide the capability to check out the safe operation of each channel of the primary flight control system and isolate any failure present in the primary flight control system to a line replaceable unit. Upon initiation, the checkout shall proceed automatically until completion or until a failure has been detected.

5.1.6.4 Maintenance. The routing checkout and isolation of failures to an LRU shall be performable by any electronic maintenance aircraft technician. The troubleshooting and repair of an LRU after removal shall be performed by designated supplier personnel.

5.1.7 Test Support. Because the DEL is part of a research and development aircraft, it shall lend itself to changes of circuit parameters during both ground and flight tests with minimal time loss. One day elapsed time for any change and checkout after change shall be a goal in the packaging design. No decaded parameters shall be acceptable. All changes of parameters shall be realized by hard-wiring and reliable workmanship in accordance with applicable military standards.

5.2 ACTUATORS

This section establishes the performance, design, and development for the servo actuator assembly (SAA). The design requirements are applicable to both the aerodynamic surface and the swashplate servo actuators. The testing requirements will be limited to accomplishment of airworthiness substantiation. Closed loop performance of the actuator, as part of the Direct Electrical Linkage System (DELS), is specified in Section 5.1.

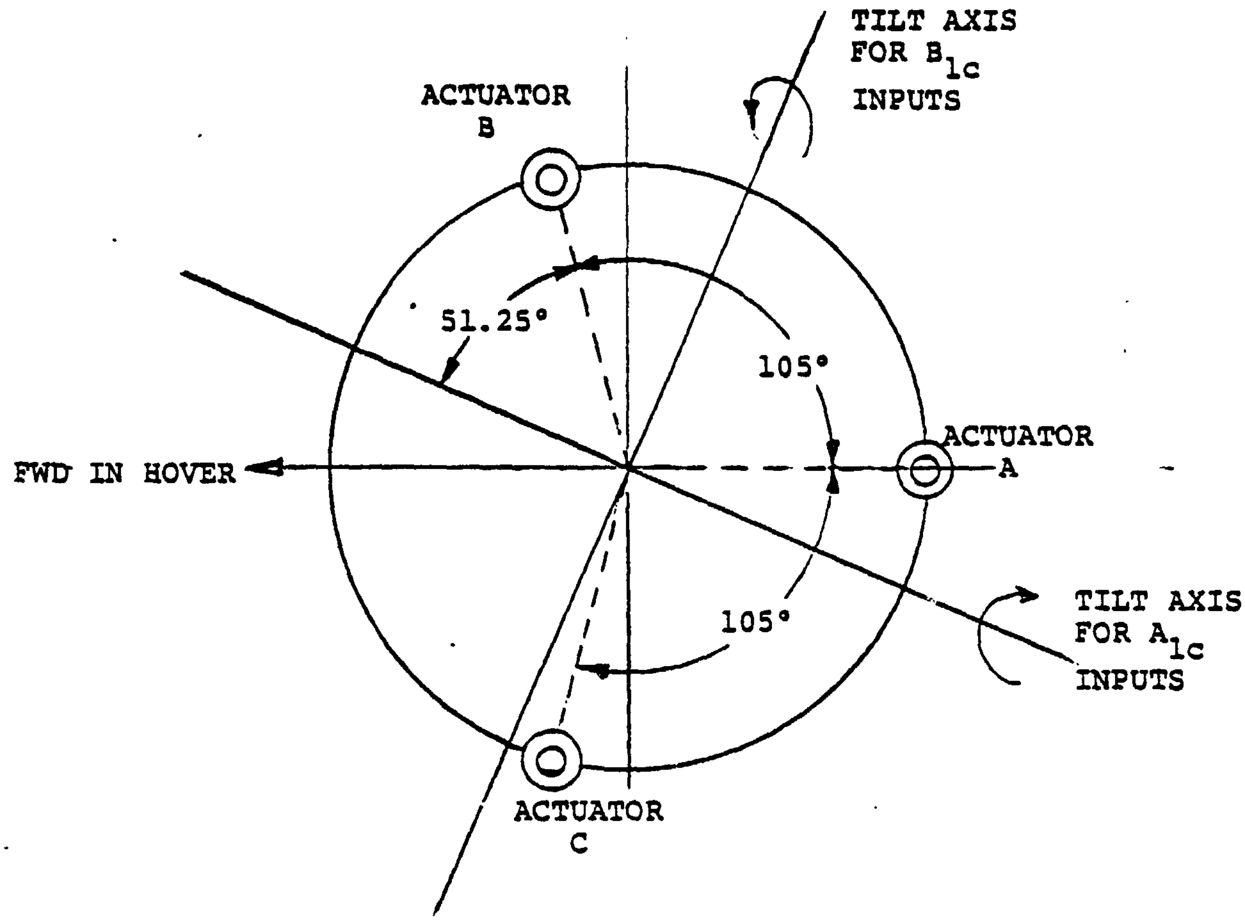
5.2.1 End Item Usage. The servo actuator will be used as the output element of the DELS.

5.2.2 Item Description. Flight control system power actuation consists of ten electrohydraulic actuators for control of the rotors and the aerodynamic surfaces. The force feel and pitch trim actuators shall be the same as are used in the mechanical flight control system of the XV-15.

The Servo Actuator Assembly (SAA) shall consist of an electro-hydraulic control stage actuator integrally manifolded to a dual system power stage actuator with a single mechanical output. This assembly converts electrical signals from the DELS control unit into mechanical output motions.

5.2.3 Description. Three servo actuator assemblies are located at each rotor head as shown in Figure A3.9. Each rotor head is independently provided with two Type II hydraulic systems

Typical Each Rotor



KINEMATICS

WITH 30° COLLECTIVE PITCH	ACTUATOR - INCHES		
	A	B	C
FOR +1° A_{1c}	RET. .0494	RET. .0957	EXT. .1212
FOR +1° B_{1c}	EXT. .1095	RET. .0768	EXT. .0187
FOR +1° COLLECTIVE PITCH	EXT. .0828 INCHES		

FIGURE A3.9. ACTUATOR LOCATION/KINEMATICS while a third Type II system is common to both heads. One independent system supplies half of the dual power stage of each SAA; the second system supplies the other half. The third system can be selected by the pilot to backup either channel. Its engagement is also conditioned by DELS logic.

All six rotor control actuators are identical. Detailed differences in piston area, manifolding and possibly electrical connections will be allowed for the aerodynamic surface control actuators. Two actuators are used for the flaperons and one each for the elevator and rudder.

5.2.4 Design. The envelope of the actuator shall not exceed the dimensions given in Figure A3.10. If possible, the thickness of the actuator should be reduced. Ideally, the actuator should fit in an annulus whose inner and outer radii are 10 inches and 14.5 inches, respectively. Two candidate cofigurations have been considered. A dual driver/dual boost design is described in the following text and is considered more desirable. A triple driver/dual boost configuration may be required to meet the reliability and maintainability requirements.

Servo loop performance and functional hardware descriptions follow for the dual driver/dual boost design. Note that the envelope requirements are defined in Figure A3.10. Similar design practices and techniques should be used for either the dual driver/dual boost or the triple driver/dual boost.

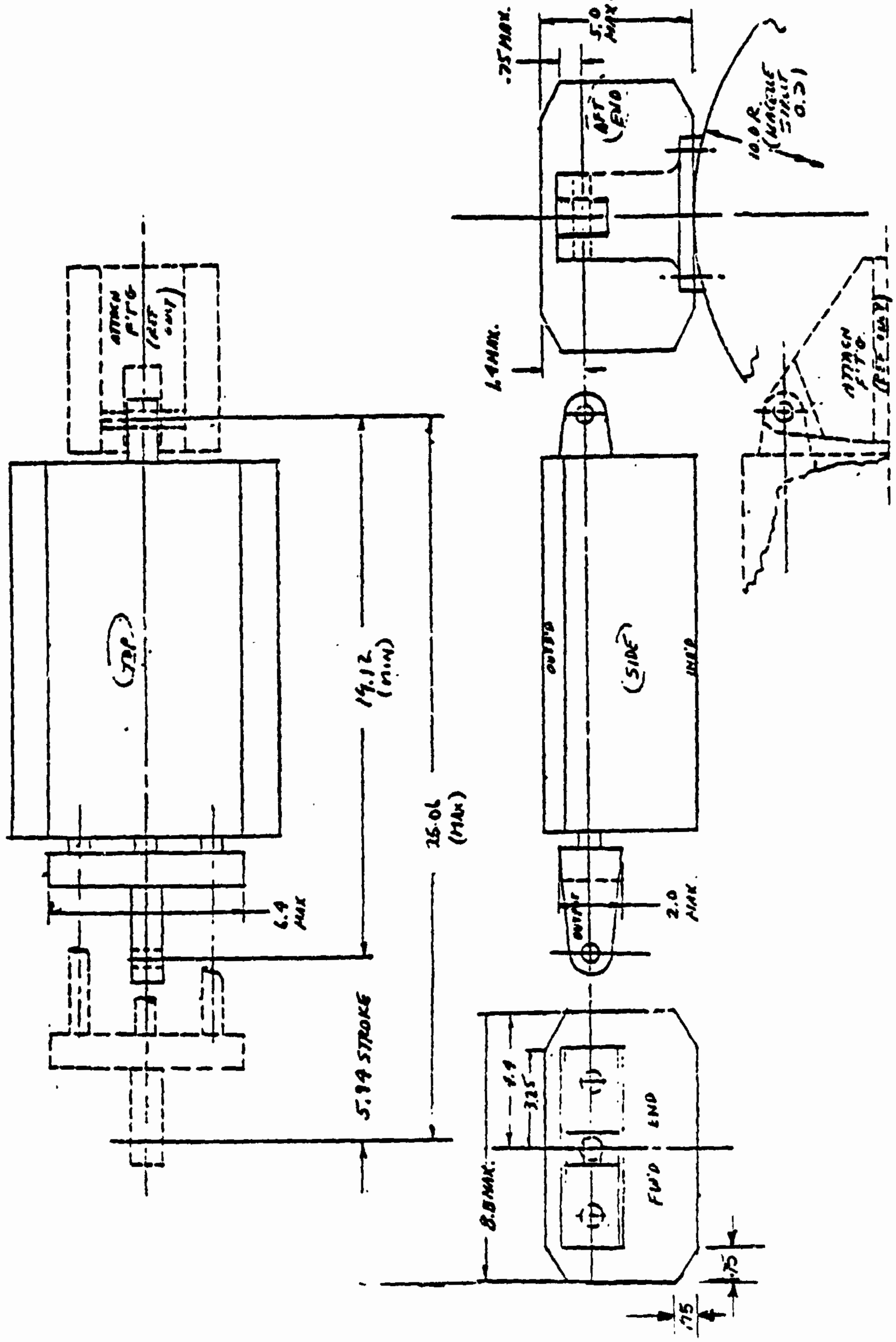


FIGURE A3.10. SERVO ACTUATOR ASSEMBLY ENVELOPE

5.2.5 Servo Performance. Figure A3.11 is a block diagram of the servo actuator servo loop. When the actuator is at rest, the control state piston and transducer are null; the power stage position transducer matches the actuator command. When the actuator command changes, the control stage piston assumes a position proportional to the generated error. This causes the power stage valve and piston movement, which reduces the servo amplifier error to zero, and the actuator is again at rest. The two-stage design effectively decouples control and power stages so that the redundancy management can be handled in the control stage where the rotor loads are not reflected and cannot upset the redundancy management.

5.2.6 Actuator Functional Description. Figure A3.12 is a block diagram of the servo actuator. Servo amplifier current positions the jet pipe of the single-stage electrohydraulic valve (EHV) which produces pressure and flow proportional to current. The EHV flow moves the control stage piston which, in turn, positions the power valve via an anti-jam bungee and force summing link. Power stage output velocity is proportional to power stage valve position. Linear variable differential transformers (LVDT) are provided to measure power and control stage piston positions. These transducers close the loops as discussed in the previous paragraph. Differential pressure transducers measure the control stage piston force which serve as a control stage performance monitor. The transducer detects high friction or jam conditions in the

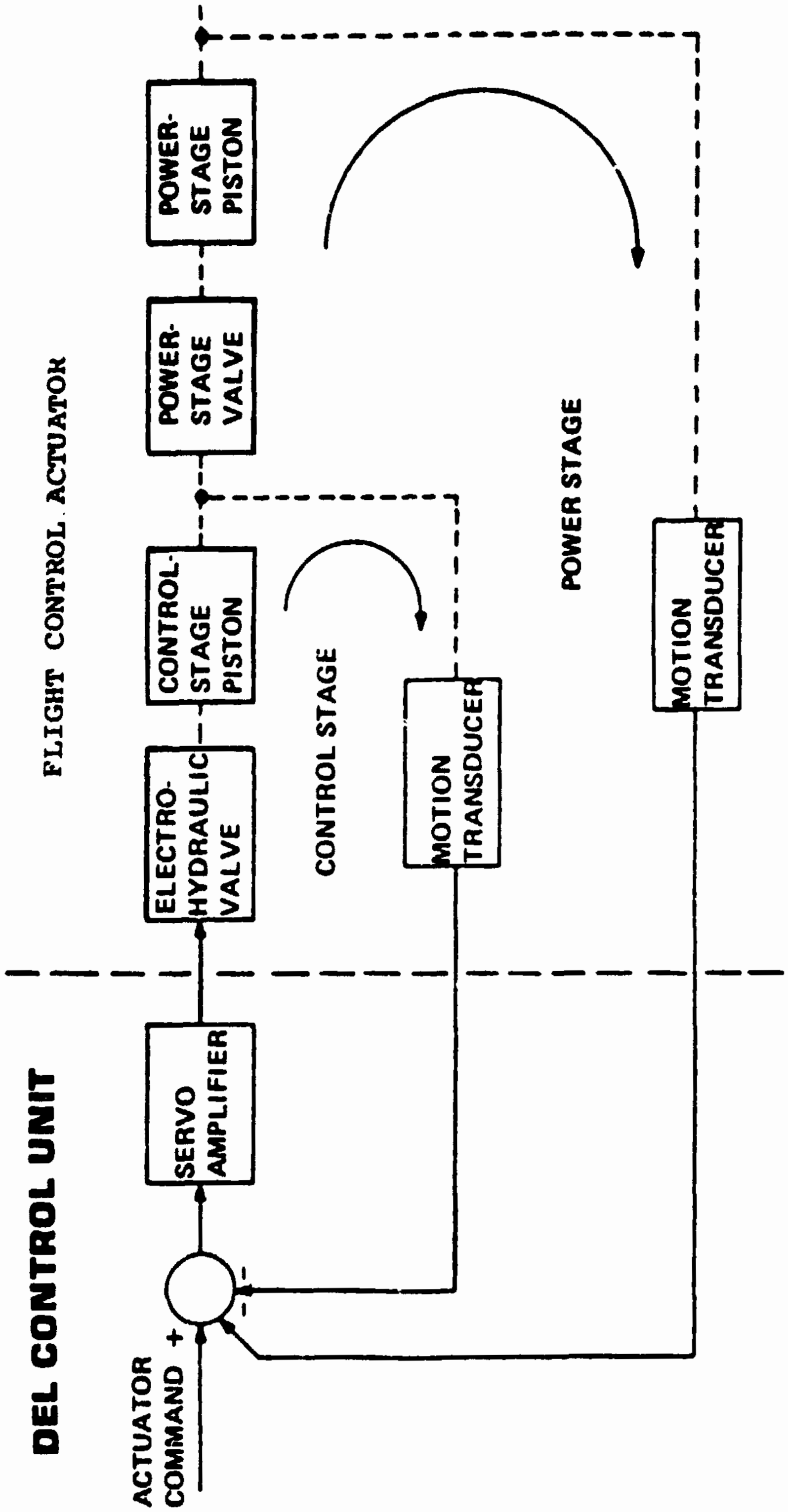


FIGURE A3.11. FLIGHT CONTROL ACTUATOR SERVOLOOP control stage pistons, intermediate linkage, or power stage valve. Sufficient overtravel is provided in the control stage piston and anti-jam bungee to allow full recovery from a control stage jam at the full power stage valve displacement. The only change under these conditions will be a 50 percent reduction in control stage loop gain. The system will be designed to tolerate this condition with no change in power loop stability.

5.2.7 Functional Interface. The functional interface between the SAA, the DELS control unit, the electrical and hydraulic power supplies, and the mechanical output are shown in block diagram form in Figure A3.12. Each of the DELS control units will provide an electrical current signal to each electro-hydraulic valve (current summing). The LVDTs of the actuators will provide information to its respective control unit about the actuator piston position, servo-valves, and differential pressure sensor.

5.2.8 Performance. The servo actuator shall be configured to meet the following requirements. Methods used to limit performance shall be approved by Boeing Vertol. It shall be a design objective to minimize the cost of producing the two configurations; the vendor should propose alternative methods for Boeing's review.

5.2.8.1 Rotor Control Function

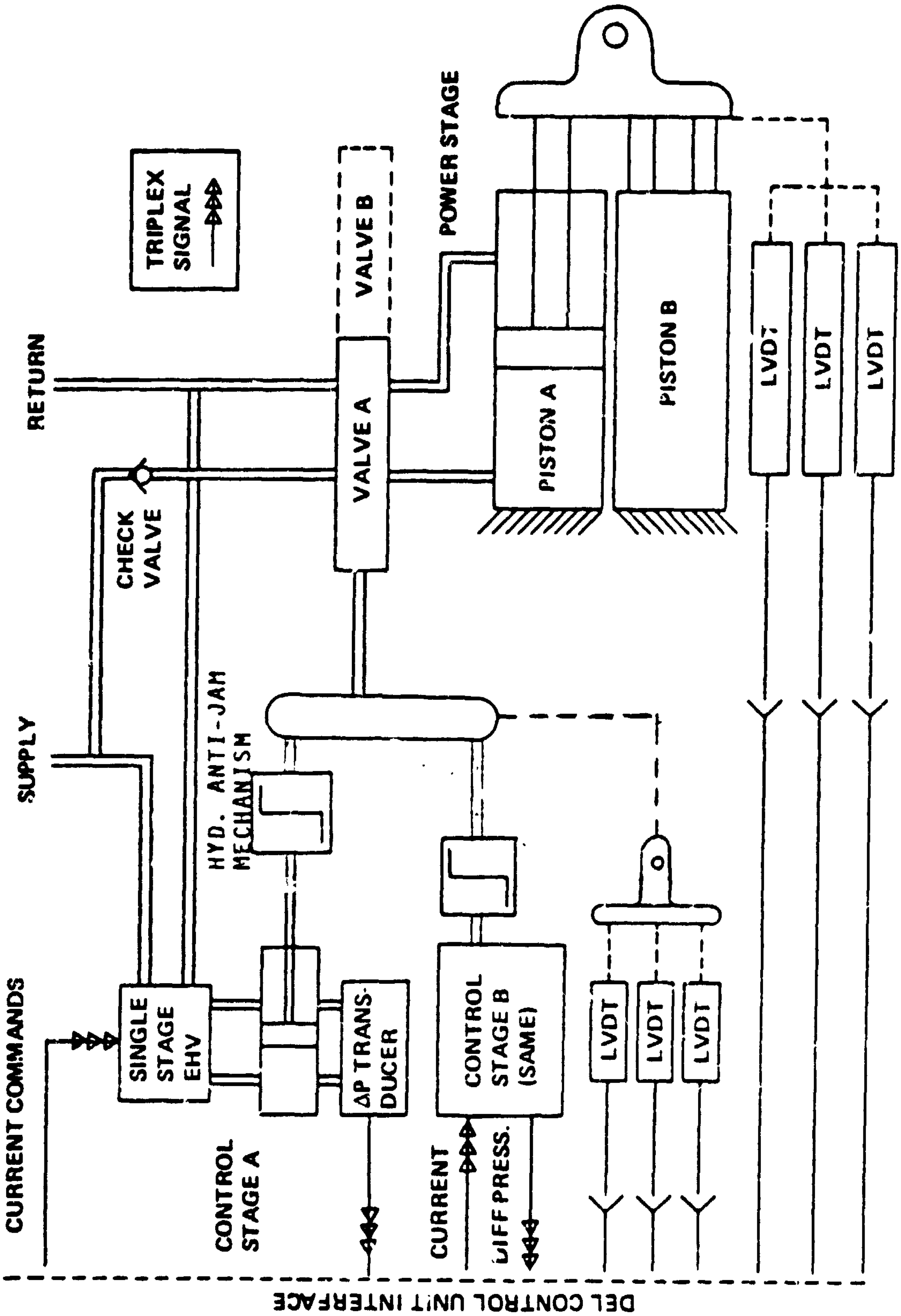


FIGURE A3.12. BLOCK DIAGRAM OF MAIN ROTOR ACTUATOR

Stroke ±3.00 inches

No Load Velocity: ± 2.5 in./sec

Fatigue Load: -1,049 lbs ± 960 lbs

Limit Load: -2,589 lbs (compressive)

Stall Force (single system): ± 2,600 lbs

Stall Force: ± 5,200 lbs

5.2.8.2 Aerodynamic Surface Function

Stroke: ± 2.00 inches

No Load Velocity: ± 5.94 in./sec

Stall Force (single system): ± 915 lbs

Stall Force: ± 1,820 lbs

5.2.8.3 Stroke Limits. The actuator shall have the capability of being driven at full speed into the fully extended or fully retracted position without external stops.

5.2.9 Environmental Conditions. The SAA shall meet the requirements of this specification during and/or following exposure to any combination of the environmental conditions described below.

5.2.9.1 Altitude. Operation without degradation of performance throughout a pressure altitude range of -200 to +20,000 feet ASL per MIL-STD-810B.

5.2.9.2 Ambient Temperature. Operation throughout an ambient temperature range of -65 to +160 degrees F.

5.2.9.3 Temperature Shock. Sudden changes in temperature of the surrounding atmosphere per MIL-STD-810B.

5.2.9.4 Humidity. Operation in a warm, highly humid atmosphere such as encountered in tropical areas per MIL-STD-810B.

5.2.9.5 Salt Fog. Operation in an atmosphere containing salt laden moisture per MIL-STD-810B.

5.2.9.6 Sand and Dust. Operation in a dust (fine sand) laden atmosphere per MIL-STD-810B.

5.2.9.7 Rain. Operation in a rain environment per MIL-STD-810B.

5.2.9.8 Immersion. Operation after immersion in hydraulic fluid at a temperature of +275° per MIL-C-5503.

5.2.9.9 Vibration. Operation during exposure to dynamic vibration stresses represented by those tests of MIL-STD-810B, Method 514.1, Procedure I, Part 1, Equipment Category (A), to include:

a. Resonance search

b. Resonance dwell

c. Cycling

5.2.9.10 Mechanical Shock. Operation after exposure to a mechanical shock environment similar to that expected in handling, transportation, and service use per MIL-STD-810B.

5.2.10 Reliability. The swashplate actuator shall be capable of meeting reliability requirements as follows.

5.2.10.1 The swashplate servoactuator, exlcuding trunnion and output rod end, shall exhibit a flight safety reliability of .99999999623, a mission reliability of .99877, and a maintenance malfunction reliability of .978 for a flight of two hours duration. Feasibility demonstration of this requirement shall consist of analytical predictions utilizing the techniques described in this section.

5.2.10.2 A single servoactuator includes cylinders, servovalves, and direct auxiliary hardware required to provide control motion to a single swashplate servoactuator position. Specifically excluded are electrical and hydraulic power supplies and control/servo electronics.

5.2.10.3 In order to provide a complete data package necessary for proper evaluation, separate models and predictions shall be generated for the following reliability objectives:

a. For reliability computations, a flight safety loss is defined as a failure which results in loss of an SAA function or damage to other aircraft equipment by actuator malfunctions (e.g., actuator on fire but still operating is considered a flight safety loss).

b. Mission abort reliability (whenever a failure occurs such that a subsequent failure could cause a flight safety loss, a mission abort is required).

c. Maintenance malfunction reliability (any failure which requires a maintenance action, regardless of functional effect, is a maintenance malfunction).

5.2.11 Maintainability. The SAA shall be designed for LRU replacement at the flight line. Routine chackout of the DELS shall be conducted using the DELS failure status/BITE panels.

5.2.11.1 Interchangeability. Interchangeability per MIL-I-8500 shall exist between all units and replaceable assemblies, subassemblies, and parts for all equipment delivered on this contract. (Not applicable to detail parts of matched assemblies).

5.2.11.2 Ground Support Equipment. Routine daily maintenance of the SAA shall be accomplished with standard hand tools available in the U.S. Army General Aircraft Mechanic's Tool Kit. No special tools or support equipment shall be required for work performed at organizational and direct support level maintenance.

5.2.11.3 Maintainability Requirements. The following maintainability requirements shall be incorporated:

a. The SAA shall be interchangeable as an LRU (line replaceable unit).

b. Servoactuators shall be removed for maintenance only "on condition". No scheduled removals shall be required.

c. Scheduled visual inspection intervals shall be no less than ten flight hours.

d. No servicing shall be necessary between inspection periods.

e. Servoactuator nameplate shall be displayed at a location as defined in envelope drawing.

f. servoactuators shall be prerigged; with no calibration requirement following installation.

g. The SAA shall incorporate suitable handling points to permit attachment to sling to raise and lower the assembly.

h. The drawing number requirements of MIL-D-1000 shall govern changes in manufacturer's part numbers.

i. The SAA rod and bearing and/or assembly shall be field replaceable.

5.3 ROTOR SPEED SENSOR

Rotor speed sensing shall be accomplished by proximity switches located in the transmission gearing. The switch shall provide 40 pulses per gear revolution, which is equivalent to 174.2 pulses per rotor revolution. The pulse characteristics such as amplitude and width and source impedance shall be determined during the detail design pahse of the system.

The number of rotor speed sensors required shall be determined by the redundancy level chosen in accordance with Section 5.1.3 of this appendix.

5.4 NACELLE INCIDENCE SENSOR

The nacelle incidence sensor shall be a synchro excited from the control unit internal A/C supply and providing an output proportional to the sine of the nacelle incidence angle. The control unit output shall be adapted to drive the existing nacelle position display and asymmetry detection system.

5.5 ENGINE N_1 CONTROL ACTUATOR

Provides control of engine power turbine in response to signals from power lever and thrust management portion of the primary flight control system. Use of existing actuator is desired. CH-47C actuator per Boeing Vertol Specification D8-2501 is a candidate.

6.0 STABILITY AND CONTROL AUGMENTATION SYSTEM (SCAS)

6.1 DESCRIPTION

The SCAS provides short and long term aircraft stabilization about the pitch, roll, and yaw axes and augmentation of cockpit control inputs to enhance aircraft maneuverability. It also provides for gust alleviation inputs to the primary system.

6.1.1 Longitudinal SCAS. The longitudinal SCAS transfer block diagram is shown in Figure A3.13. Pitch rate and pitch attitude are programmed as functions of airspeed to provide longitudinal stability. Cockpit control quickening in the longitudinal axis is also provided.

6.1.2 Lateral SCAS. The lateral SCAS transfer block diagram is shown in Figure A3.14. Roll rate, roll attitude, and sideslip are the parameters sensed and processed to provide lateral stabilization. Cockpit control quickening in the lateral axis is provided at low airspeeds.

6.1.3 Directional SCAS. The directional SCAS transfer block diagram is provided in Figure A3.15. Yaw rate, yaw attitude, and sideslip are the parameters used for stabilization. Turn coordination and roll into yaw cross coupling operation are also prohibited through the processing of roll bank angle and roll rate. Cockpit control quickening in the yaw axis is also provided.

6.1.4 Logic. The logic controlling the lateran and directional SCAS functions is shown in Figure A3.16.

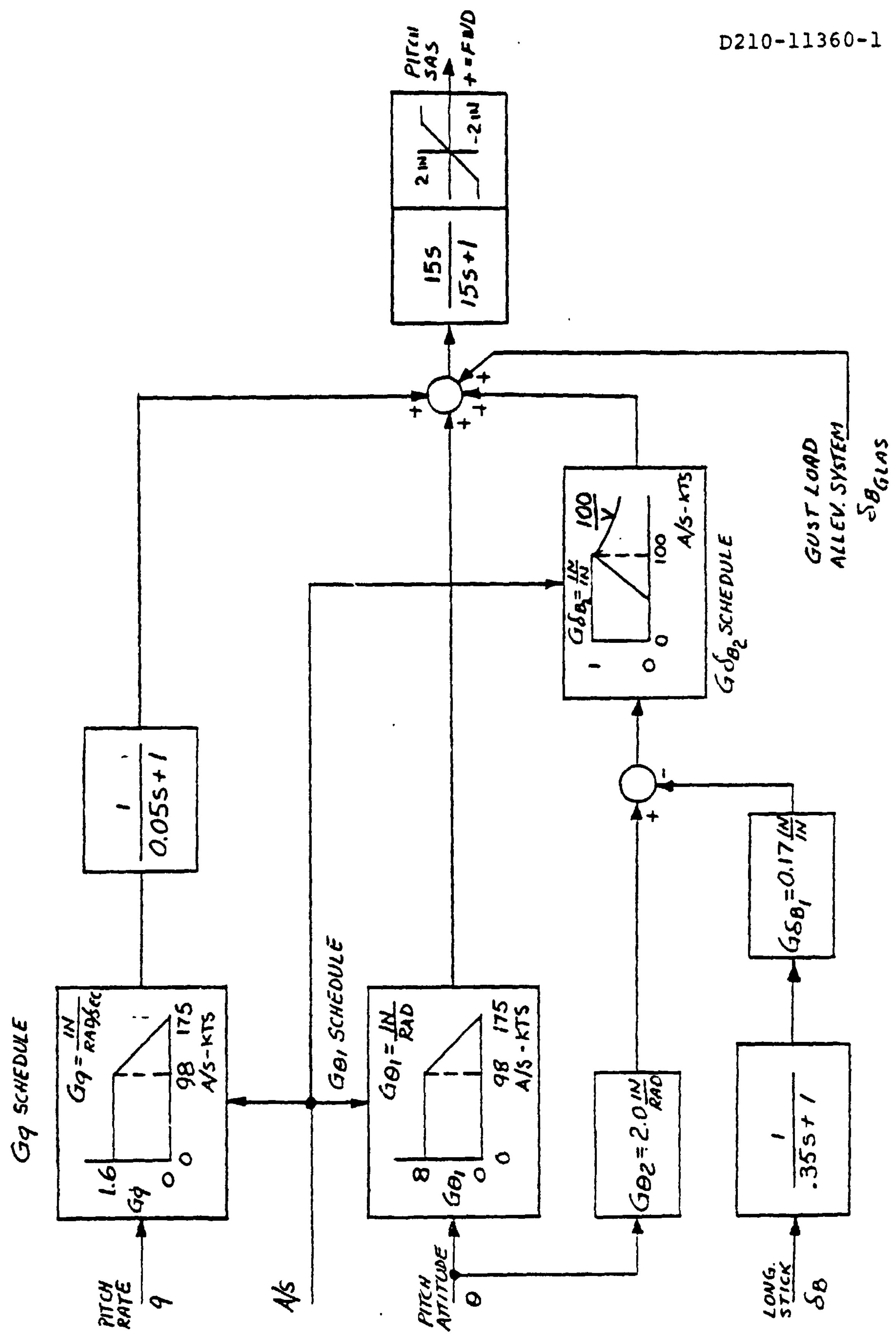


FIGURE A3.13. LONGITUDINAL SCAS

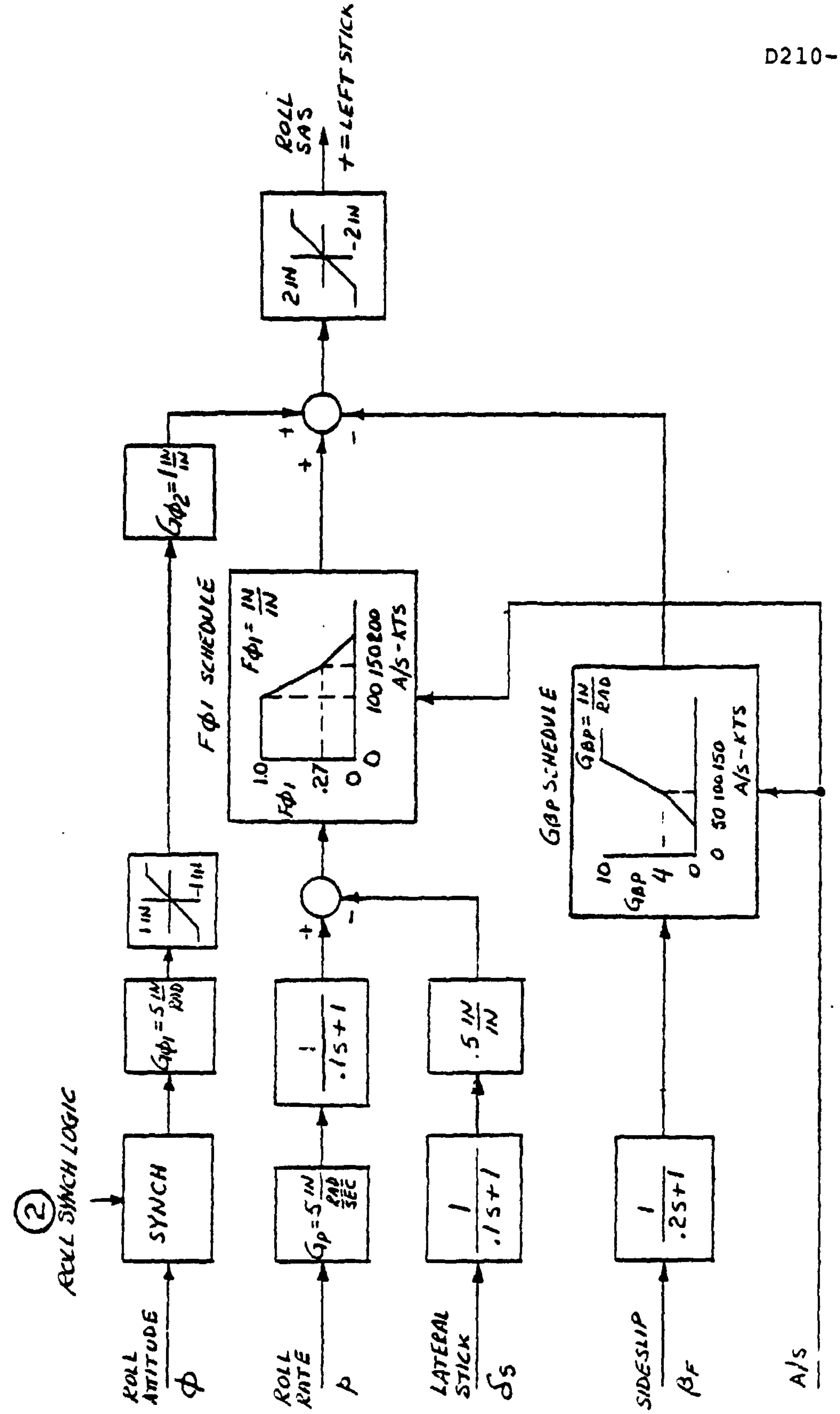


FIGURE A3.14. LATERAL SCAS

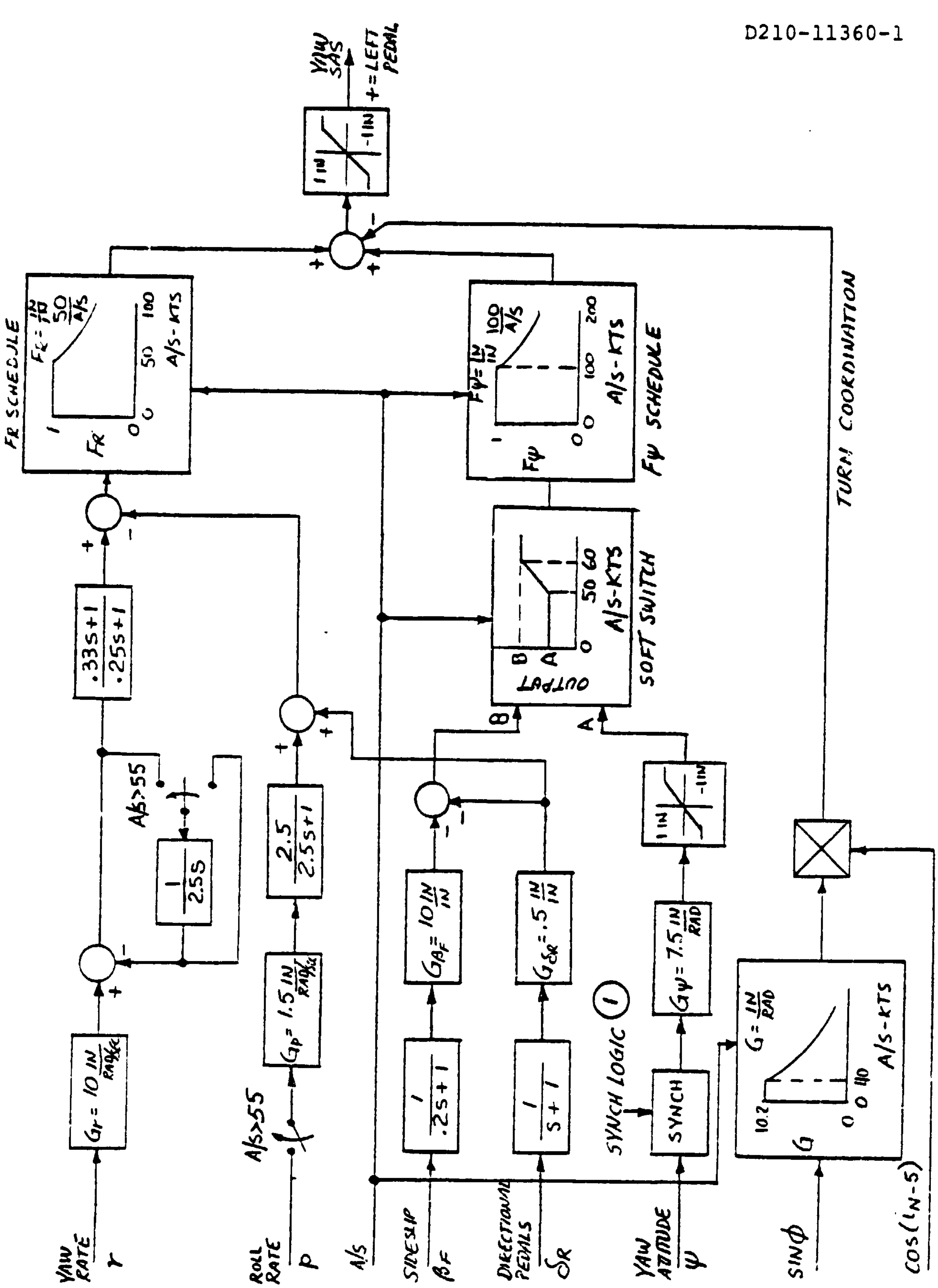


FIGURE A3.15. DIRECTIONAL SCAS

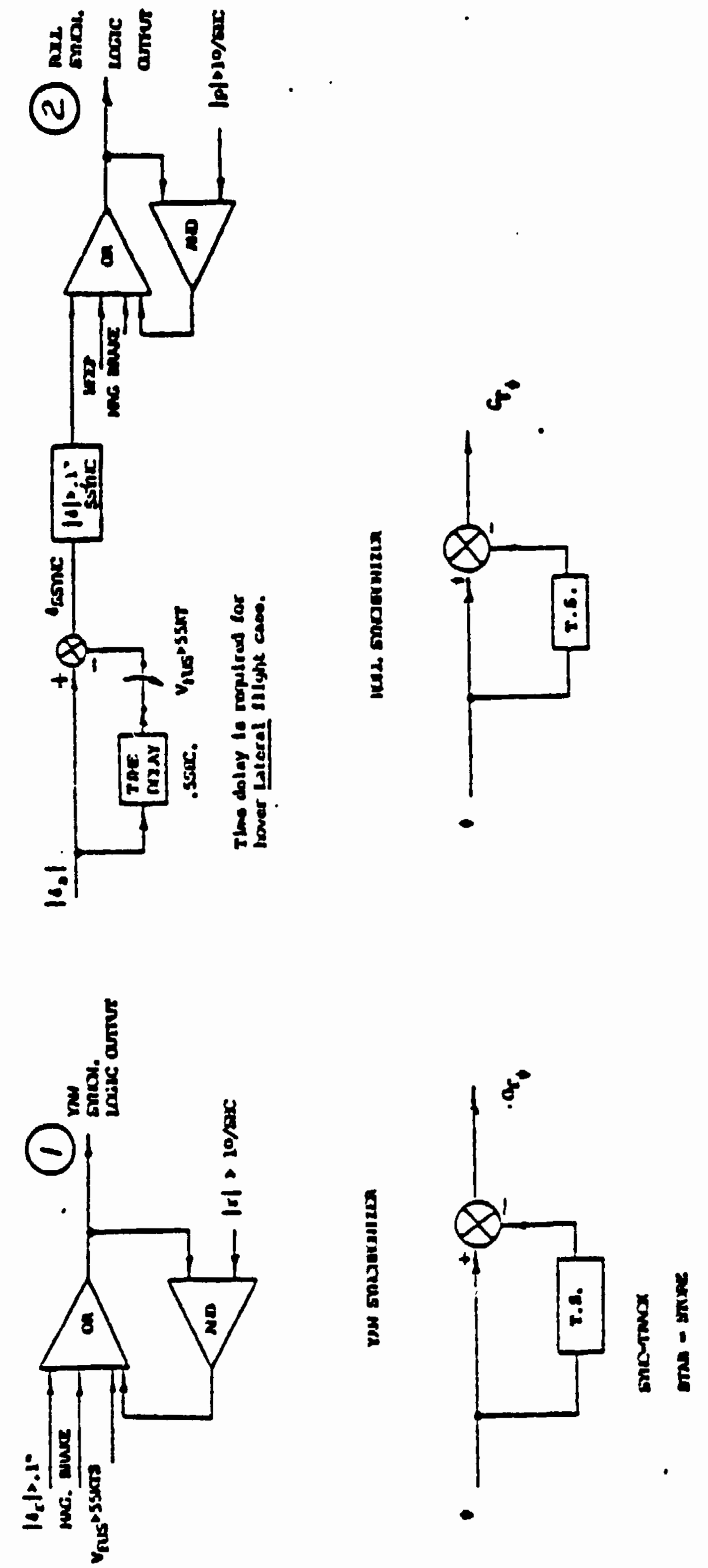


FIGURE A3.16. LATERAL DIRECTIONAL SCAS - SYNCHRONIZER AND LOGIC

6.1.5 Gust Alleviation. System sensing and transfer functions are being defined. Sensing could be vertical acceleration, pitch acceleration, or angle of attack. Sensor signal to be processed through a filter such as shown in Figure A3.17 and input to flaps and/or elevator. Assume SCAS unit incorporates three such filters.

6.2 SYSTEM PERFORMANCE

6.2.1 System Accuracies. System gain, schedule, and transfer function accuracies shall be met over the range of environments defined in Paragraph 3.4.

6.2.2 Steady State Accuracy. The steady state accuracy of SCAS shall be 5 percent or better. For a given control input, the accuracy is defined as the percentage difference between the desired actuator command voltage and the actual command voltage. The above accuracies include the schedule accuracies.

6.2.3 Null Accuracy. The total steady state null accuracy associated with the SCAS from sensor inputs to actuator command shall be .5 percent full scale.

6.2.4 Resolution. Resolution is defined as the minimum change in control input required to obtain actuator command change. The resolution (equated to actuator command change) shall not exceed .1 percent full scale.

6.2.5 System Hysteresis. Hysteresis within the SCAS shall not exceed .1 percent full scale.

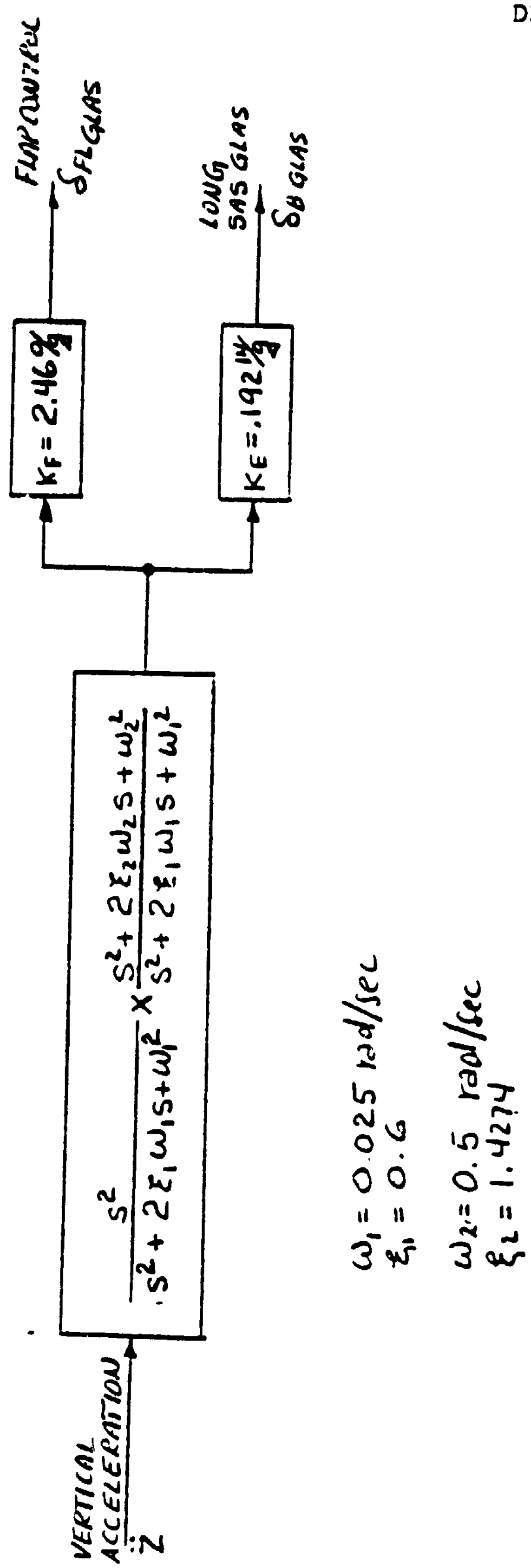


FIGURE A3.17. TYPICAL GUST ALLEVIATION CONFIGURATION

6.2.6 Frequency Response. The linear range frequency response of each of the transfer paths (sensor input to actuator command) shall be flat to within ± 1 db and within ±10° phase shift to 5 radians per second where no filtering is required. Where filtering is required, the frequency response gain shall be within ± 2 db and phase shift shall be within ± 10° of the theoretical value.

6.3 REDUNDANCY

a. The SCAS shall be at least single fail operative for any failure.

b. All failures causing loss of one SCAS channel shall be detected and immediately displayed.

c. The allowable transient due to a failure shall meet the requirements of Paragraph 3.1.4.

6.4 RELIABILITY

The overall system reliability of the SCAS shall be .999 for a two-hour flight per SCAS channel. Reliability shall be demonstrated by analytical methods based on known failure rates of components used in the design. The required redundancy level shall be adopted to meet this reliability requirement.

6.5 IMPLEMENTATION OPTIONS

The following options shall be suitable for the design of the SCAS electronics.

a. Analog signal processing.

b. Digital signal processing.

c. A combination of analog and digital signal processing.

The option can be also exercised if combining PFCS signal processing and SCAS processing in the same circuit axes or maintaining separation of the PFCS and SCAS electronics.

7.0 REFERENCE SPECIFICATIONS AND DOCUMENTS

FAR xx	Tentative Airworthiness Standards of Powered Lift Transport Category Aircraft, August 1970.
NASA CR-151950	Preliminary Simulation of an Advanced Hingeless Rotor XV-15 Tilt-Rotor Aircraft, Boeing Vertol, December 1976.
MIL-HDBK-217A	Reliability Stress and Failure Rate Data for Electronic Equipment.
MIL-E-5400R	Electronic Equipment, General Specifications for.
MIL-STD-810B	Environmental Test Methods.
MIL-H-5440E	Hydraulic Systems Aircraft Types 1 and 2, Design, Installation, and Data Requirements for.
MIL-C-5503C	Cylinders, Aeronautical, Hydraulics, Actuating, General Requirement for.
MIL-H-8775C	Hydraulic System Components, Aircraft and Missiles, General Specification for.
D8-2501-1	Procurement Specification, Proportional N_1 Engine Control System.

APPENDIX IV

NASTRAN MODEL COMPUTER INPUT

FOR

STRUCTURAL ANALYSIS

APPENDIX IV. NORMAL MODES ANALYSIS INPUT DATA

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
1-	CBAR	278	1	73	92	82			2	
2-	CBAR	281	1	76	95	79			2	
3-	CBAR	282	1	77	96	78			2	
4-	CBAR	283	1	78	97	77			2	
5-	CBAR	284	1	79	98	76			2	
6-	CBAR	287	1	82	101	73			2	
7-	CBAR	314	4	90	109	91			2	
8-	CBAR	315	4	109	115	108			2	
9-	CBAR	316	4	115	123	114			2	
10-	CBAR	317	4	123	135	122			2	
11-	CBAR	318	4	108	109	114			2	
12-	CBAR	319	4	114	115	122			2	
13-	CBAR	320	4	122	123	115			2	
14-	CBAR	321	4	91	108	90			2	
15-	CBAR	322	4	108	114	109			2	
16-	CBAR	323	4	114	122	115			2	
17-	CBAR	324	4	122	133	123			2	
18-	CBAR	327	6	123	124	116			2	
19-	CBAR	331	4	121	132	124			2	
20-	CBAR	334	4	121	124	116			2	
21-	CBAR	338	4	124	136	132			2	
22-	CBAR	341	6	122	120	113			2	
23-	CBAR	347	6	124	125	137			2	
24-	CBAR	350	6	122	121	114			2	
25-	CBAR	353	4	120	125	112			2	
26-	CBAR	357	6	125	126	141			2	
27-	CBAR	358	6	120	119	112			2	
28-	CBAR	359	4	119	126	141			2	
29-	CBAR	360	4	126	141	130			2	
30-	CBAR	361	6	126	127	142			2	
31-	CBAR	362	6	119	118	130			2	
32-	CBAR	363	4	119	130	141			2	
33-	CBAR	364	4	118	127	129			2	
34-	CBAR	365	15	143	144	156			2	
35-	CBAR	366	15	143	156	144			2	
36-	CBAR	367	15	143	157	158			2	
37-	CBAR	368	15	143	158	157			2	
38-	CBAR	369	15	157	158	143			2	
39-	CBAR	370	15	157	156	168			2	
40-	CBAR	371	15	158	144	168			2	

TILT ROTOR NORMAL MODES ANALYSIS
SYMMETRIC MODES CRUISE ATTITUDE

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S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
41-	CBAR	372	4	89	149	150			2	
42-	CBAR	373	4	89	150	149			2	
43-	CBAR	374	4	89	152	148			2	
44-	CBAR	375	4	89	148	152			2	
45-	CBAR	376	4	149	150	89			2	
46-	CBAR	377	4	150	152	148			2	
47-	CBAR	378	4	149	148	152			2	
48-	CBAR	379	2	89	188	196			2	
49-	CBAR	380	2	88	186	188			2	
50-	CBAR	381	2	89	201	199			2	
51-	CBAR	382	2	88	199	201			2	
52-	CBAR	383	20	152	163	230			2	
53-	CBAR	384	20	153	163	230			2	
54-	CBAR	385	20	163	164	230			2	
55-	CBAR	386	20	154	164	230			2	
56-	CBAR	387	20	164	165	230			2	
57-	CBAR	388	20	162	163	230			2	
58-	CBAR	389	20	151	162	230			2	
59-	CBAR	390	20	162	165	230			2	
60-	CBAR	391	20	165	166	230			2	
61-	CBAR	392	20	155	167	230			2	
62-	CBAR	393	20	156	167	230			2	
63-	CBAR	394	20	156	167	230			2	
64-	CBAR	395	20	166	168	230			2	
65-	CBAR	396	20	159	166	230			2	
66-	CBAR	397	20	160	165	230			2	
67-	CBAR	398	20	161	162	230			2	
68-	CBAR	399	20	148	161	230			2	
69-	CBAR	400	20	160	161	230			2	
70-	CBAR	401	20	159	160	230			2	
71-	CBAR	402	20	144	159	230			2	
72-	CBAR	403	20	145	159	230			2	
73-	CBAR	404	20	146	160	230			2	
74-	CBAR	405	20	147	161	230			2	
75-	CBAR	466	15	157	180	169			2	
76-	CBAR	467	15	157	169	180			2	
77-	CBAR	468	15	158	169	170			2	
78-	CBAR	469	15	158	170	169			2	
79-	CBAR	470	4	150	176	175			2	
80-	CBAR	471	4	150	175	176			2	

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
81-	CBAR	472	4	149	175	150			2	
82-	CBAR	473	4	149	174	175			2	
83-	CBAR	474	20	164	167	230			2	
84-	CBAR	475	5	150	151	152			2	
85-	CBAR	476	5	157	168	156			2	
86-	CBAR	477	5	149	151	150			2	
87-	CBAR	478	5	158	168	157			2	
88-	CBAR	781	6	127	244	118			2	
89-	CBAR	782	6	118	245	127			2	
90-	CBAR	783	6	244	246	245			2	
91-	CBAR	784	6	245	247	244			2	
92-	CBAR	785	6	246	103	247			2	
93-	CBAR	786	6	247	104	246			2	
94-	CBAR	787	16	248	250	138			2	
95-	CBAR	788	16	248	140	137			2	&C788
96-	&C788		456							
97-	CBAR	789	5	250	138	140			2	&C789
98-	&C789	245								
99-	CBAR	790	16	248	249	251			2	
100-	CBAR	791	16	249	251	248			2	
101-	CBAR	792	5	251	128	249			2	&C792
102-	&C792	345	5							
103-	CBAR	793	18	1	252	3			2	
104-	CBAR	794	18	2	252	3			2	
105-	CBAR	795	18	6	252	3			2	
106-	CBAR	796	18	7	252	3			2	
107-	CBAR	797	19	252	253	1			2	
108-	CBAR	800	1	84	86	79			2	&C1
109-	&C1	456	4							
110-	CBAR	801	2	86	87	97			2	
111-	CBAR	802	3	86	97	87			2	
112-	CBAR	803	3	86	95	87			2	
113-	CBAR	804	3	86	96	87			2	
114-	CBAR	805	3	86	98	87			2	
115-	CBAR	806	4	97	96	78			2	
116-	CBAR	807	4	98	95	79			2	
117-	CBAR	808	5	99	94	80			2	
118-	CBAR	809	5	100	93	81			2	
119-	CBAR	810	5	101	92	82			2	
120-	CBAR	811	5	90	91	109			2	

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
121-	CBAR	812	5	102	72	110			2	
122-	CBAR	813	5	103	104	111			2	
123-	CBAR	814	6	97	98	78			2	
124-	CBAR	815	6	98	99	79			2	
125-	CBAR	816	7	99	100	80			2	
126-	CBAR	817	7	100	101	81			2	
127-	CBAR	818	6	96	95	77			2	
128-	CBAR	819	6	95	94	76			2	
129-	CBAR	820	7	94	93	75			2	
130-	CBAR	821	7	93	92	74			2	
131-	CBAR	822	8	85	92	89			2	
132-	CBAR	823	8	85	101	89			2	
133-	CBAR	824	8	85	90	89			2	
134-	CBAR	825	8	85	91	89			2	
135-	CBAR	826	8	90	102	109			2	
136-	CBAR	827	8	102	103	110			2	
137-	CBAR	828	8	72	104	107			2	
138-	CBAR	829	8	91	72	108			2	
139-	CBAR	830	9	83	85	82			2	&C7
140-	&C7	456	4							
141-	CBAR	831	10	85	89	101			2	&C8
142-	&C8		4							
143-	CBAR	832	5	143	134	135			2	&C9
144-	&C9	1456								
145-	CBAR	833	14	134	135	143			2	
146-	CBAR	834	14	134	133	143			2	
147-	CBAR	835	4	135	133	123			2	
148-	CBAR	836	4	136	132	124			2	
149-	CBAR	837	4	137	131	125			2	
150-	CBAR	838	4	141	130	126			2	
151-	CBAR	839	4	142	129	127			2	
152-	CBAR	840	6	129	130	118			2	
153-	CBAR	841	6	130	131	119			2	
154-	CBAR	842	6	131	132	120			2	
155-	CBAR	843	6	132	133	121			2	
156-	CBAR	844	6	136	135	124			2	
157-	CBAR	845	6	137	136	125			2	
158-	CBAR	846	6	141	137	126			2	
159-	CBAR	847	6	142	141	127			2	
160-	CBAR	848	14	137	138	136			2	

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
161-	CBAR	849	14	139	138	140			2	
162-	CBAR	850	14	137	139	138			2	
163-	CBAR	851	14	139	140	138			2	
164-	CBAR	852	14	139	131	140			2	
165-	CBAR	853	14	125	137	120			2	
166-	CBAR	854	14	131	140	139			2	
167-	CBAR	855	14	128	142	127			2	
168-	CBAR	856	14	129	128	142			2	
169-	CBAR	857	14	127	142	119			2	
170-	CBAR	858	11	229	217	230			2	&C13
171-	&C13		456							
172-	CBAR	859	11	229	219	230			2	&C14
173-	&C14		456							
174-	CBAR	860	11	229	221	230			2	&C15
175-	&C15		456							
176-	CBAR	861	11	229	223	230			2	&C16
177-	&C16		456							
178-	CBAR	862	11	229	225	230			2	&C17
179-	&C17		456							
180-	CBAR	863	11	229	227	230			2	&C18
181-	&C18		456							
182-	CBAR	864	14	217	228	229			2	
183-	CBAR	865	14	217	218	229			2	
184-	CBAR	866	14	218	219	229			2	
185-	CBAR	867	14	219	220	229			2	
186-	CBAR	868	14	220	221	229			2	
187-	CBAR	869	14	221	222	229			2	
188-	CBAR	870	14	222	223	229			2	
189-	CBAR	871	14	223	224	229			2	
190-	CBAR	872	14	224	225	229			2	
191-	CBAR	873	14	225	226	229			2	
192-	CBAR	874	14	226	227	229			2	
193-	CBAR	875	14	227	228	229			2	
194-	CBAR	876	12	201	202	195			2	
195-	CBAR	877	12	202	203	196			2	
196-	CBAR	878	12	203	204	197			2	
197-	CBAR	879	12	193	204	199			2	
198-	CBAR	880	12	193	194	200			2	
199-	CBAR	881	12	194	195	201			2	
200-	CBAR	882	12	195	196	202			2	

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	1	2	3	4	5	6	7	8	9	10
201-	CBAR	883	12	196	197	203			2	
202-	CBAR	884	12	197	198	204			2	
203-	CBAR	885	12	198	199	193			2	
204-	CBAR	886	12	199	200	194			2	
205-	CBAR	887	12	200	201	194			2	
206-	CBAR	888	12	187	188	181			2	
207-	CBAR	889	12	188	189	192			2	
208-	CBAR	890	12	189	190	183			2	
209-	CBAR	891	12	190	191	194			2	
210-	CBAR	892	12	191	192	196			2	
211-	CBAR	893	12	181	192	187			2	
212-	CBAR	894	12	181	182	198			2	
213-	CBAR	895	12	182	183	189			2	
214-	CBAR	896	12	193	184	189			2	
215-	CBAR	897	12	194	185	191			2	
216-	CBAR	898	12	195	186	192			2	
217-	CBAR	899	12	196	187	192			2	
218-	CBAR	900	15	212	213	206			2	
219-	CBAR	901	15	213	214	207			2	
220-	CBAR	902	15	214	215	209			2	
221-	CBAR	903	15	215	216	209			2	
222-	CBAR	904	15	205	216	210			2	
223-	CBAR	905	15	205	206	211			2	
224-	CBAR	906	15	206	207	212			2	
225-	CBAR	907	15	207	208	214			2	
226-	CBAR	908	15	208	209	214			2	
227-	CBAR	909	15	209	210	216			2	
228-	CBAR	910	15	210	211	205			2	
229-	CBAR	911	15	211	212	206			2	
230-	CBAR	912	15	172	173	178			2	
231-	CBAR	913	15	173	174	179			2	
232-	CBAR	914	15	174	175	180			2	
233-	CBAR	915	15	175	176	169			2	
234-	CBAR	916	15	176	177	170			2	
235-	CBAR	917	15	177	178	171			2	
236-	CBAR	918	15	178	179	173			2	
237-	CBAR	919	15	179	180	175			2	
238-	CBAR	920	15	169	180	175			2	
239-	CBAR	921	15	169	170	176			2	
240-	CBAR	922	15	170	171	177			2	

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
241-	CBAR	923	13	171	172	177			2	
242-	CBAR	924	14	145	146	154			2	
243-	CBAR	925	14	146	147	154			2	
244-	CBAR	926	14	147	148	154			2	
245-	CBAR	927	14	148	151	155			2	
246-	CBAR	928	14	151	152	158			2	
247-	CBAR	929	14	152	153	168			2	
248-	CBAR	930	14	153	154	144			2	
249-	CBAR	931	14	154	155	146			2	
250-	CBAR	932	14	155	156	147			2	
251-	CBAR	933	14	156	168	151			2	
252-	CBAR	934	14	144	168	153			2	
253-	CBAR	935	14	144	145	152			2	
254-	CBAR	936	16	228	230	217			2	
255-	CBAR	937	14	120	131	125			2	
256-	CBAR	938	14	118	129	127			2	
257-	CBAR	939	17	101	90	105			2	
258-	CBAR	940	17	92	91	108			2	
259-	CBAR	941	18	3	252	2			2	
260-	CBAR	942	18	4	252	2			2	
261-	CBAR	943	18	5	252	2			2	
262-	CBAR	944	18	8	252	2			2	
263-	CBAR	945	18	9	252	2			2	
264-	CBAR	946	18	10	252	2			2	
265-	CBAR	1201	14	90	101	91			2	
266-	CBAR	1202	14	91	92	90			2	
267-	CONM2	1100	1	0	33.47					
268-	CONM2	1101	2	0	33.47					
269-	CONM2	1102	6	0	33.47					
270-	CONM2	1103	7	0	33.47					
271-	CONM2	1104	11	0	25.72					
272-	CONM2	1105	12	0	25.72					
273-	CONM2	1106	16	0	25.72					
274-	CONM2	1107	17	0	25.72					
275-	CONM2	1108	21	0	41.56					
276-	CONM2	1109	22	0	41.56					
277-	CONM2	1110	26	1	41.56					
278-	CONM2	1111	27	1	41.56					
279-	CONM2	1112	31	1	36.62					
280-	CONM2	1113	32	1	36.62					

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
281-	CONM2	1114	36	1	36.62					
282-	CONM2	1115	37	1	36.62					
283-	CONM2	1116	41	1	70.97					
284-	CONM2	1117	42	1	70.97					
285-	CONM2	1118	46	1	70.97					
286-	CONM2	1119	47	1	70.97					
287-	CONM2	1120	51	1	29.72					
288-	CONM2	1121	52	1	29.72					
289-	CONM2	1122	56	1	29.72					
290-	CONM2	1123	57	1	29.72					
291-	CONM2	1124	61	1	36.15					
292-	CONM2	1125	62	1	36.15					
293-	CONM2	1126	66	1	36.15					
294-	CONM2	1127	67	1	36.15					
295-	CONM2	1128	92	1	15.97					
296-	CONM2	1129	96	1	15.97					
297-	CONM2	1130	97	1	15.97					
298-	CONM2	1131	101	1	15.97					
299-	CONM2	1132	230	0	252.0					
300-	CONM2	1133	229	2	401.31					&2
301-	&2	150142.0		150527.0			17412.0			
302-	CONM2	1134	89	2	75.84					&3
303-	&3	28374.0		28447.0			3290.0			
304-	CONM2	1135	143	2	75.84					&4
305-	&4	28374.0		28447.0			3298.0			
306-	CONM2	1136	252	0	3233.0					&5
307-	&5			2.35E7			2.36E7			
306-	CONM2	1138	248	0	776.0					&7
309-	&7	99691.0		266616.0			187790.0			
310-	CONROD	100	1	2	1	1.2				
311-	CONROD	101	2	3	1	1.2				
312-	CONROD	102	1	10	1	1.2				
313-	CONROD	103	3	10	1	.04				
314-	CONROD	104	3	4	1	1.2				
315-	CONROD	105	9	10	1	1.2				
316-	CONROD	106	4	9	1	.04				
317-	CONROD	107	4	5	1	1.2				
318-	CONROD	108	8	9	1	1.2				
319-	CONROD	109	5	8	1	.04				
320-	CONROD	110	5	6	1	1.2				

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
321-	CONROD	111	7	8	1	1.2				
322-	CONROD	112	6	7	1	1.2				
323-	CONROD	113	2	12	1	1.2				
324-	CONROD	114	3	13	1	1.12				
325-	CONROD	115	4	14	1	1.14				
326-	CONROD	116	5	15	1	1.19				
327-	CONROD	117	6	16	1	1.2				
328-	CONROD	118	7	17	1	1.2				
329-	CONROD	119	8	18	1	1.19				
330-	CONROD	120	9	19	1	1.14				
331-	CONROD	121	10	20	1	1.12				
332-	CONROD	122	1	11	1	1.2				
333-	CONROD	123	11	12	1	1.2				
334-	CONROD	124	12	13	1	1.2				
335-	CONROD	125	13	14	1	1.2				
336-	CONROD	126	14	15	1	1.2				
337-	CONROD	127	15	16	1	1.2				
338-	CONROD	128	16	17	1	1.2				
339-	CONROD	129	17	18	1	1.2				
340-	CONROD	130	15	18	1	.04				
341-	CONROD	131	18	19	1	1.2				
342-	CONROD	132	14	19	1	.04				
343-	CONROD	133	19	20	1	1.2				
344-	CONROD	134	13	20	1	.04				
345-	CONROD	135	11	20	1	1.2				
346-	CONROD	136	11	21	1	1.2				
347-	CONROD	137	12	22	1	1.2				
348-	CONROD	138	13	23	1	1.12				
349-	CONROD	139	14	24	1	1.14				
350-	CONROD	140	15	25	1	1.19				
351-	CONROD	141	16	26	1	1.2				
352-	CONROD	142	17	27	1	1.2				
353-	CONROD	143	18	28	1	1.19				
354-	CONROD	144	19	29	1	1.14				
355-	CONROD	145	20	30	1	1.12				
356-	CONROD	146	21	22	1	1.2				
357-	CONROD	147	22	23	1	1.2				
358-	CONROD	148	23	24	1	1.2				
359-	CONROD	149	24	25	1	1.2				
360-	CONROD	150	25	26	1	1.2				

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
361-	CONROD	151	26	27	1	1.2				
362-	CONROD	152	27	28	1	1.2				
363-	CONROD	153	28	29	1	1.2				
364-	CONROD	154	29	30	1	1.2				
365-	CONROD	155	21	30	1	1.2				
366-	CONROD	156	21	31	1	1.2				
367-	CONROD	157	22	32	1	1.2				
368-	CONROD	158	23	33	1	1.12				
369-	CONROD	159	24	34	1	1.14				
370-	CONROD	160	25	35	1	1.19				
371-	CONROD	161	26	36	1	1.2				
372-	CONROD	162	27	37	1	1.2				
373-	CONROD	163	28	38	1	1.19				
374-	CONROD	154	29	39	1	1.14				
375-	CONROD	165	30	40	1	1.12				
376-	CONROD	166	31	32	1	1.2				
377-	CONROD	167	32	33	1	1.2				
378-	CONROD	168	33	34	1	1.2				
379-	CONROD	169	34	35	1	1.2				
380-	CONROD	170	35	36	1	1.2				
381-	CONROD	171	36	37	1	1.2				
382-	CONROD	172	37	38	1	1.2				
383-	CONROD	173	38	39	1	1.2				
384-	CONROD	174	39	40	1	1.2				
385-	CONROD	175	31	40	1	1.2				
386-	CONROD	176	31	41	1	1.2				
387-	CONROD	177	32	42	1	1.2				
388-	CONROD	178	33	43	1	1.12				
389-	CONROD	179	34	44	1	1.14				
390-	CONROD	180	35	45	1	1.19				
391-	CONROD	181	36	46	1	1.2				
392-	CONROD	182	37	47	1	1.2				
393-	CONROD	183	38	48	1	1.19				
394-	CONROD	184	39	49	1	1.14				
395-	CONROD	185	40	50	1	1.12				
396-	CONROD	186	41	42	1	1.2				
397-	CONROD	187	42	43	1	1.2				
398-	CONROD	188	43	44	1	1.2				
399-	CONROD	189	44	45	1	1.2				
400-	CONROD	190	45	46	1	1.2				

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 .. 8 .. 9 .. 10 .
401-	CONROD	191	46	47	1	1.2	
402-	CONROD	192	47	48	1	1.2	
403-	CONROD	193	45	48	1	.04	
404-	CONROD	194	48	49	1	1.2	
405-	CONROD	195	44	49	1	.04	
406-	CONROD	196	49	50	1	1.2	
407-	CONROD	197	43	50	1	.04	
408-	CONROD	198	41	50	1	1.2	
409-	CONROD	199	41	51	1	1.2	
410-	CONROD	200	42	[illegible]	1	1.2	
411-	CONROD	201	43	53	1	1.12	
412-	CONROD	202	44	54	1	1.14	
413-	CONROD	203	45	55	1	1.19	
414-	CONROD	204	46	56	1	1.2	
415-	CONROD	205	47	57	1	1.2	
416-	CONROD	206	48	58	1	1.19	
417-	CONROD	207	49	59	1	1.14	
418-	CONROD	208	50	60	1	1.12	
419-	CONROD	209	51	52	1	1.2	
420-	CONROD	210	52	53	1	1.2	
421-	CONROD	211	53	54	1	1.2	
422-	CONROD	212	54	55	1	1.2	
423-	CONROD	213	55	56	1	1.2	
424-	CONROD	214	56	57	1	1.2	
425-	CONROD	215	57	58	1	1.2	
426-	CONROD	216	58	59	1	1.2	
427-	CONROD	217	59	60	1	1.2	
428-	CONROD	218	51	60	1	1.2	
429-	CONROD	219	51	61	1	1.2	
430-	CONROD	220	52	62	1	1.2	
431-	CONROD	221	53	63	1	1.12	
432-	CONROD	222	54	64	1	1.14	
433-	CONROD	223	55	65	1	1.19	
434-	CONROD	224	56	66	1	1.2	
435-	CONROD	225	57	67	1	1.2	
436-	CONROD	226	58	68	1	1.19	
437-	CONROD	227	59	69	1	1.14	
438-	CONROD	228	60	70	1	1.12	
439-	CONROD	229	61	62	1	1.2	
440-	CONROD	230	62	63	1	1.2	

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
441-	CONROD	231	63	64	1	1.2				
442-	CONROD	232	64	65	1	1.2				
443-	CONROD	233	65	66	1	1.2				
444-	CONROD	234	66	67	1	1.2				
445-	CONROD	235	67	68	1	1.2				
446-	CONROD	236	65	68	1	.94				
447-	CONROD	237	68	69	1	1.2				
448-	CONROD	238	64	69	1	.84				
449-	CONROD	239	59	70	1	1.2				
450-	CONROD	240	63	70	1	.84				
451-	CONROD	241	61	70	1	1.2				
452-	CONROD	242	61	72	1	1.2				
453-	CONROD	243	62	73	1	1.2				
454-	CONROD	244	63	74	1	1.12				
455-	CONROD	245	59	75	1	1.14				
456-	CONROD	246	65	76	1	1.19				
457-	CONROD	247	66	77	1	1.2				
458-	CONROD	248	67	78	1	1.2				
459-	CONROD	249	68	79	1	1.19				
460-	CONROD	250	69	80	1	1.14				
461-	CONROD	251	70	81	1	1.12				
462-	CONROD	254	73	74	1	1.2				
463-	CONROD	255	74	75	1	1.2				
464-	CONROD	256	75	76	1	1.2				
465-	CONROD	257	76	77	1	1.2				
466-	CONROD	258	77	78	1	1.2				
467-	CONROD	259	78	79	1	1.2				
468-	CONROD	260	76	79	1	.06				
469-	CONROD	261	79	80	1	1.2				
470-	CONROD	262	75	80	1	.66				
471-	CONROD	263	80	81	1	1.2				
472-	CONROD	264	74	81	1	.06				
473-	CONROD	265	81	82	1	1.2				
474-	CONROD	266	73	82	1	1.2				
475-	CONROD	270	73	83	1	3.				
476-	CONROD	271	82	83	1	3.				
477-	CONROD	272	79	84	1	.35				
478-	CONROD	273	76	84	1	.35				
479-	CONROD	274	77	84	1	.35				
480-	CONROD	275	78	84	1	.35				

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S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
481-	CONROD	279	76	93	1	1.17				
482-	CONROD	280	75	94	1	1.19				
483-	CONROD	285	80	99	1	1.19				
484-	CONROD	286	81	100	1	1.17				
485-	CONROD	304	87	88	2	.65				
486-	CONROD	305	23	30	1	.01				
487-	CONROD	306	24	29	1	.01				
488-	CONROD	307	25	24	1	.01				
489-	CONROD	308	33	40	1	.01				
490-	CONROD	309	34	39	1	.01				
491-	CONROD	310	35	38	1	.01				
492-	CONROD	311	53	60	1	.01				
493-	CONROD	312	54	59	1	.01				
494-	CONROD	313	55	58	1	.01				
495-	CONROD	325	109	110	1	.30				
496-	CONROD	326	115	116	1	.30				
497-	CONROD	328	72	107	1	.35				
498-	CONROD	329	107	113	1	.35				
499-	CONROD	330	113	121	1	.35				
500-	CONROD	332	107	110	1	.25				
501-	CONROD	333	113	116	1	.25				
502-	CONROD	335	102	118	1	.35				
503-	CONROD	336	110	116	1	.35				
504-	CONROD	337	116	124	1	.35				
505-	CONROD	339	106	107	1	.3				
506-	CONROD	340	112	113	1	.3				
507-	CONROD	342	103	111	1	.35				
508-	CONROD	343	111	117	1	.35				
509-	CONROD	344	117	125	1	.35				
510-	CONROD	345	110	111	1	.30				
511-	CONROD	346	116	117	1	.30				
512-	CONROD	348	107	108	1	.30				
513-	CONROD	349	113	114	1	.30				
514-	CONROD	351	105	111	1	.35				
515-	CONROD	352	112	117	1	.35				
516-	CONROD	354	104	136	1	.35				
517-	CONROD	355	106	112	1	.35				
518-	CONROD	356	112	120	1	.35				
519-	CONROD	406	148	174	1	1.400				
520-	CONROD	407	147	173	1	.70				

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
521-	CONROD	408	146	172	1	.35				
522-	CONROD	409	145	171	1	.35				
523-	CONROD	410	144	170	1	.45				
524-	CONROD	411	168	169	1	.45				
525-	CONROD	412	156	180	1	.45				
526-	CONROD	413	155	179	1	.35				
527-	CONROD	414	154	178	1	.35				
528-	CONROD	415	153	177	1	.70				
529-	CONROD	416	152	176	1	1.40				
530-	CONROD	417	151	175	1	1.00				
531-	CONROD	418	175	188	1	.7				
532-	CONROD	419	176	189	1	.75				
533-	CONROD	420	177	190	1	.40				
534-	CONROD	421	178	191	1	.35				
535-	CONROD	422	179	192	1	.35				
536-	CONROD	423	180	181	1	.42				
537-	CONROD	424	169	182	1	.50				
538-	CONROD	425	170	183	1	.42				
539-	CONROD	426	171	184	1	.35				
540-	CONROD	427	172	185	1	.35				
541-	CONROD	428	173	166	1	.40				
542-	CONROD	429	174	187	1	.75				
543-	CONROD	430	187	200	1	.53				
544-	CONROD	431	186	199	1	.35				
545-	CONROD	432	185	198	1	.35				
546-	CONROD	433	184	197	1	.35				
547-	CONROD	434	183	196	1	.53				
548-	CONROD	435	182	195	1	.7				
549-	CONROD	436	181	194	1	.53				
550-	CONROD	437	192	193	1	.35				
551-	CONROD	438	191	204	1	.35				
552-	CONROD	439	190	203	1	.35				
553-	CONROD	440	189	202	1	.53				
554-	CONROD	441	188	201	1	.7				
555-	CONROD	442	201	214	1	.7				
556-	CONROD	443	202	215	1	.44				
557-	CONROD	444	203	216	1	.53				
558-	CONROD	445	204	205	1	.35				
559-	CONROD	446	193	206	1	.53				
560-	CONROD	447	194	207	1	.44				

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 .. 9 .. 10 .
561-	CONROD	448	195	208	1	.7		
562-	CONROD	449	136	209	1	.44		
563-	CONROD	450	197	210	1	.53		
564-	CONROD	451	198	211	1	.35		
565-	CONROD	452	199	212	1	.33		
566-	CONROD	453	200	213	1	.44		
567-	CONROD	454	213	226	1	.35		
568-	CONROD	455	212	225	1	.7		
569-	CONROD	456	211	224	1	.35		
570-	CONROD	457	210	223	1	.7		
571-	CONROD	458	209	222	1	.55		
572-	CONROD	459	208	221	1	.7		
573-	CONROD	460	207	220	1	.35		
574-	CONROD	461	206	219	1	.7		
575-	CONROD	462	205	218	1	.35		
576-	CONROD	463	216	217	1	.7		
577-	CONROD	464	215	228	1	.55		
578-	CONROD	465	214	227	1	.7		
579-	CONROD	768	111	246	1	.3		
580-	CONROD	769	246	247	1	.3		
581-	CONROD	770	106	247	1	.3		
582-	CONROD	771	126	242	1	.3		
583-	CONROD	772	119	243	1	.3		
584-	CONROD	773	242	246	1	.3		
585-	CONROD	774	243	247	1	.3		
586-	CONROD	775	242	243	1	.3		
587-	CONROD	776	117	242	1	.3		
588-	CONROD	777	112	243	1	.3		
589-	CONROD	778	242	244	1	.3		
590-	CONROD	779	243	245	1	.3		
591-	CONROD	780	244	245	1	.3		
592-	CORD1C	2	231	230	89			
593-	CORD1R	1	21	11	22			
594-	CQUAD2	726	9	163	153	154	164	
595-	CQUAD2	727	8	164	154	155	167	
596-	CQUAD2	728	8	151	152	163	162	
597-	CQUAD2	729	8	162	163	164	165	
598-	CQUAD2	730	8	165	164	167	166	
599-	CQUAD2	731	8	166	167	156	168	
600-	CQUAD2	732	8	148	151	162	161	

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
601-	CQUAD2	733	8	161	162	165	160			
602-	CQUAD2	734	8	160	165	166	159			
603-	CQUAD2	735	8	159	166	168	144			
604-	CQUAD2	736	8	147	161	160	146			
605-	CQUAD2	737	8	146	160	159	145			
606-	CSHEAR	500	1	1	2	3	10			
607-	CSHEAR	501	1	3	4	9	10			
608-	CSHEAR	502	1	4	5	8	9			
609-	CSHEAR	503	1	5	6	7	8			
610-	CSHEAR	504	2	5	6	16	15			
611-	CSHEAR	505	2	4	5	15	14			
612-	CSHEAR	506	2	3	4	14	13			
613-	CSHEAR	507	2	2	3	13	12			
614-	CSHEAR	508	1	1	2	12	11			
615-	CSHEAR	509	2	1	10	20	11			
616-	CSHEAR	510	2	10	9	19	20			
617-	CSHEAR	511	2	9	8	18	19			
618-	CSHEAR	512	2	8	7	17	18			
619-	CSHEAR	513	1	7	6	16	17			
620-	CSHEAR	514	1	15	16	17	18			
621-	CSHEAR	515	1	14	15	18	19			
622-	CSHEAR	516	1	13	14	19	20			
623-	CSHEAR	517	1	11	12	13	20			
624-	CSHEAR	518	1	11	12	22	21			
625-	CSHEAR	519	2	12	13	23	22			
626-	CSHEAR	520	2	13	14	24	23			
627-	CSHEAR	521	2	14	15	25	24			
628-	CSHEAR	522	2	15	16	26	25			
629-	CSHEAR	523	1	17	16	25	27			
630-	CSHEAR	524	2	18	17	27	28			
631-	CSHEAR	525	2	19	18	28	29			
632-	CSHEAR	526	2	20	19	29	30			
633-	CSHEAR	527	2	11	20	30	21			
634-	CSHEAR	528	3	25	26	27	28			
635-	CSHEAR	529	3	24	25	28	29			
636-	CSHEAR	530	3	23	24	29	30			
637-	CSHEAR	531	3	22	23	30	21			
638-	CSHEAR	532	1	21	22	32	31			
639-	CSHEAR	533	2	22	23	33	32			
640-	CSHEAR	534	2	23	24	34	33			

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
641-	CSHEAR	535	2	24	25	35	34			
642-	CSHEAR	536	2	25	26	36	35			
643-	CSHEAR	537	1	27	26	36	37			
644-	CSHEAR	538	2	28	27	37	38			
645-	CSHEAR	539	2	29	28	38	39			
646-	CSHEAR	540	2	30	29	39	40			
647-	CSHEAR	541	2	21	30	40	31			
648-	CSHEAR	542	3	35	36	37	38			
649-	CSHEAR	543	3	34	35	38	39			
650-	CSHEAR	544	3	33	34	39	40			
651-	CSHEAR	545	3	32	33	40	31			
652-	CSHEAR	546	1	31	32	42	41			
653-	CSHEAR	547	2	32	33	43	42			
654-	CSHEAR	548	2	33	34	44	43			
655-	CSHEAR	549	2	34	35	45	44			
656-	CSHEAR	550	2	35	36	45	45			
657-	CSHEAR	551	1	37	36	45	47			
658-	CSHEAR	552	2	38	37	47	48			
659-	CSHEAR	553	2	39	38	48	49			
660-	CSHEAR	554	2	40	39	49	50			
661-	CSHEAR	555	2	31	40	50	41			
662-	CSHEAR	556	4	45	46	47	48			
663-	CSHEAR	557	4	44	45	48	49			
664-	CSHEAR	558	4	43	44	49	50			
665-	CSHEAR	559	4	42	43	50	41			
666-	CSHEAR	560	1	41	42	52	51			
667-	CSHEAR	561	2	42	43	53	52			
668-	CSHEAR	562	2	43	44	54	53			
669-	CSHEAR	563	2	44	45	55	54			
670-	CSHEAR	564	2	45	46	55	55			
671-	CSHEAR	565	1	47	46	56	57			
672-	CSHEAR	[illegible]	2	48	47	57	58			
673-	CSHEAR		2	49	48	58	59			
674-	CSHEAR		2	50	49	59	60			
675-	CSHEAR	569	2	41	50	60	51			
676-	CSHEAR	570	3	55	56	57	58			
677-	CSHEAR	571	3	54	55	58	59			
678-	CSHEAR	572	3	53	54	59	60			
679-	CSHEAR	573	3	52	53	60	51			
680-	CSHEAR	574	1	51	52	62	61			

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
681-	CSHEAR	575	2	52	53	63	62			
682-	CSHEAR	576	2	53	54	64	63			
683-	CSHEAR	577	2	54	55	65	64			
684-	CSHEAR	578	2	55	56	66	65			
685-	CSHEAR	579	1	57	56	65	67			
686-	CSHEAR	580	2	58	57	67	68			
687-	CSHEAR	581	2	59	58	68	69			
688-	CSHEAR	582	2	60	59	69	70			
689-	CSHEAR	583	2	51	60	70	61			
690-	CSHEAR	584	4	65	66	67	68			
691-	CSHEAR	585	4	64	65	68	69			
692-	CSHEAR	586	4	63	64	69	70			
693-	CSHEAR	587	4	62	63	70	61			
694-	CSHEAR	588	1	61	62	73	82			
695-	CSHEAR	589	2	62	63	74	73			
696-	CSHEAR	590	2	63	64	75	74			
697-	CSHEAR	591	2	64	65	76	75			
698-	CSHEAR	592	2	65	66	77	76			
699-	CSHEAR	593	1	67	66	77	78			
700-	CSHEAR	594	2	68	67	78	79			
701-	CSHEAR	595	2	69	68	79	80			
702-	CSHEAR	596	2	70	69	80	81			
703-	CSHEAR	597	2	61	70	81	82			
704-	CSHEAR	598	5	75	76	79	80			
705-	CSHEAR	599	5	74	75	80	81			
706-	CSHEAR	600	5	73	74	81	82			
707-	CSHEAR	603	2	73	74	93	92			
708-	CSHEAR	604	2	74	75	94	93			
709-	CSHEAR	605	2	75	76	95	94			
710-	CSHEAR	606	2	76	77	95	95			
711-	CSHEAR	607	1	78	77	96	97			
712-	CSHEAR	608	2	79	78	97	98			
713-	CSHEAR	609	2	80	79	98	99			
714-	CSHEAR	610	2	81	80	99	100			
715-	CSHEAR	611	2	82	81	100	101			
716-	CSHEAR	613	5	94	95	98	99			
717-	CSHEAR	614	5	93	94	99	100			
718-	CSHEAR	615	5	92	93	100	101			
719-	CSHEAR	616	1	82	73	92	101			
720-	CSHEAR	617	1	79	76	95	98			

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
721-	CSHEAR	618	1	91	90	102	72			
722-	CSHEAR	619	1	72	102	103	104			
723-	CSHEAR	620	5	108	109	110	107			
724-	CSHEAR	621	5	107	110	111	106			
725-	CSHEAR	622	5	114	115	116	113			
726-	CSHEAR	623	5	113	116	117	112			
727-	CSHEAR	624	8	122	123	124	121			
728-	CSHEAR	625	8	121	124	125	120			
729-	CSHEAR	626	7	120	125	126	119			
730-	CSHEAR	627	7	119	126	127	118			
731-	CSHEAR	628	1	133	135	136	132			
732-	CSHEAR	629	1	132	136	137	131			
733-	CSHEAR	630	1	131	137	141	130			
734-	CSHEAR	631	1	130	141	142	129			
735-	CSHEAR	632	6	91	90	109	108			
736-	CSHEAR	633	6	108	109	115	114			
737-	CSHEAR	634	6	114	115	123	122			
738-	CSHEAR	635	6	122	123	135	133			
739-	CSHEAR	636	8	72	102	110	107			
740-	CSHEAR	637	8	107	110	116	113			
741-	CSHEAR	638	8	113	116	124	121			
742-	CSHEAR	639	8	121	124	136	132			
743-	CSHEAR	640	7	104	103	111	105			
744-	CSHEAR	641	7	106	111	117	112			
745-	CSHEAR	642	7	112	117	125	120			
746-	CSHEAR	643	7	120	125	137	131			
747-	CSHEAR	644	5	119	126	141	130			
748-	CSHEAR	645	7	118	127	142	129			
749-	CSHEAR	646	8	72	91	108	107			
750-	CSHEAR	647	8	107	108	114	113			
751-	CSHEAR	648	8	113	114	122	121			
752-	CSHEAR	649	8	121	122	133	132			
753-	CSHEAR	650	8	104	72	107	106			
754-	CSHEAR	651	8	106	107	113	112			
755-	CSHEAR	652	8	112	113	121	120			
756-	CSHEAR	653	8	120	121	132	131			
757-	CSHEAR	654	8	119	120	131	130			
758-	CSHEAR	655	8	118	119	130	129			
759-	CSHEAR	656	8	102	90	109	110			
760-	CSHEAR	657	8	110	109	115	116			

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
761-	CSHEAR	658	8	116	115	123	124			
762-	CSHEAR	659	8	124	123	135	136			
763-	CSHEAR	660	8	103	102	110	111			
764-	CSHEAR	661	8	111	110	116	117			
765-	CSHEAR	662	8	117	116	124	125			
766-	CSHEAR	663	8	125	124	136	137			
767-	CSHEAR	664	8	126	125	137	141			
768-	CSHEAR	665	8	127	126	141	142			
769-	CSHEAR	666	7	148	174	173	147			
770-	CSHEAR	667	7	147	173	172	146			
771-	CSHEAR	668	7	146	172	171	145			
772-	CSHEAR	669	7	145	171	170	144			
773-	CSHEAR	670	6	144	170	169	168			
774-	CSHEAR	671	6	168	169	180	156			
775-	CSHEAR	672	7	156	180	179	155			
776-	CSHEAR	673	7	155	179	178	154			
777-	CSHEAR	674	7	154	178	177	153			
778-	CSHEAR	675	7	153	177	176	152			
779-	CSHEAR	676	6	152	176	175	151			
780-	CSHEAR	677	6	151	175	174	148			
781-	CSHEAR	678	6	175	188	187	174			
782-	CSHEAR	679	6	176	189	188	175			
783-	CSHEAR	680	7	177	190	189	176			
784-	CSHEAR	681	7	178	191	190	177			
785-	CSHEAR	682	7	179	192	191	178			
786-	CSHEAR	683	7	180	181	192	179			
787-	CSHEAR	684	6	169	182	181	180			
788-	CSHEAR	685	6	170	183	182	169			
789-	CSHEAR	686	7	171	184	183	170			
790-	CSHEAR	687	7	172	185	184	171			
791-	CSHEAR	688	7	173	186	185	172			
792-	CSHEAR	689	7	174	187	186	173			
793-	CSHEAR	690	7	187	200	199	186			
794-	CSHEAR	691	7	186	199	198	185			
795-	CSHEAR	692	7	185	198	197	184			
796-	CSHEAR	693	7	194	197	196	183			
797-	CSHEAR	694	6	183	196	195	182			
798-	CSHEAR	695	6	182	195	194	181			
799-	CSHEAR	696	7	181	194	193	192			
800-	CSHEAR	697	7	192	193	204	191			

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 .. 9 .. 10 .
801-	CSHEAR	698	7	191	204	203	190	
802-	CSHEAR	699	7	190	203	202	189	
803-	CSHEAR	700	6	189	202	201	188	
804-	CSHEAR	701	6	188	201	200	187	
805-	CSHEAR	702	6	201	214	213	200	
806-	CSHEAR	703	6	202	215	214	201	
807-	CSHEAR	704	7	203	216	215	202	
808-	CSHEAR	705	7	204	205	216	203	
809-	CSHEAR	706	7	193	206	205	204	
810-	CSHEAR	707	7	194	207	206	193	
811-	CSHEAR	708	6	195	208	207	194	
812-	CSHEAR	709	6	196	209	208	195	
813-	CSHEAR	710	7	197	210	209	196	
814-	CSHEAR	711	7	198	211	210	197	
815-	CSHEAR	712	7	199	212	211	198	
816-	CSHEAR	713	7	200	213	212	199	
817-	CSHEAR	714	6	213	226	225	212	
818-	CSHEAR	715	6	212	225	224	211	
819-	CSHEAR	716	6	211	224	223	210	
820-	CSHEAR	717	6	210	223	222	209	
821-	CSHEAR	718	6	209	222	221	208	
822-	CSHEAR	719	6	208	221	220	207	
823-	CSHEAR	720	6	207	220	219	206	
824-	CSHEAR	721	6	206	219	218	205	
825-	CSHEAR	722	6	205	218	217	216	
826-	CSHEAR	723	6	216	217	228	215	
827-	CSHEAR	724	6	215	228	227	214	
828-	CSHEAR	725	6	214	227	226	213	
829-	CSHEAR	754	8	111	117	242	246	
830-	CSHEAR	755	8	117	125	126	242	
831-	CSHEAR	756	8	242	126	127	244	
832-	CSHEAR	757	8	106	112	243	247	
833-	CSHEAR	758	8	112	120	119	243	
834-	CSHEAR	759	8	243	119	118	245	
835-	CSHEAR	760	8	246	242	243	247	
836-	CSHEAR	761	8	242	126	119	243	
837-	CSHEAR	762	8	103	246	247	104	
838-	CSHEAR	763	8	246	244	245	247	
839-	CSHEAR	764	8	244	127	118	245	
840-	CSHEAR	765	8	117	242	243	112	

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
841-	CSHEAR	766	8	242	244	245	243			
842-	CSHEAR	767	8	111	246	247	106			
843-	CTRIA2	750	1	103	111	246				
844-	CTRIA2	751	1	246	242	244				
845-	CTRIA2	752	1	104	106	247				
846-	CTRIA2	753	1	247	243	245				
847-	CTRIA2	1000	1	163	153	152				
848-	CTRIA2	1001	1	167	156	155				
849-	CTRIA2	1002	1	161	148	147				
850-	CTRIA2	1003	1	159	145	144				
851-	EIGR	1	INV	.0	12.	4	0	0	.001	&1
852-	&1	MAX								
853-	GRDSET		1				0	456		
854-	GRID	1	0	315.	0.0	101.02650		0		
855-	GRID	2	0	315.	0.0	87.6585 0		0		
856-	GRID	3	0	308.	0.0	87.407850		0		
857-	GRID	4	0	301.	0.0	86.906550		0		
858-	GRID	5	0	294.	0.0	89.243350		0		
859-	GRID	5	0	287.	0.0	91.2095 0		0		
860-	GRID	7	0	287.	0.0	97.4755 0		0		
861-	GRID	8	0	294.	0.0	100.44170		0		
862-	GRID	9	0	301.	0.0	101.77850		0		
863-	GRID	10	0	308.	0.0	101.27720		0		
864-	GRID	11	0	315.	28.	101.02650				
865-	GRID	12	0	315.	28.	87.6585 0				
866-	GRID	13	0	308.	28.	87.407850				
867-	GRID	14	0	301.	28.	86.906550				
868-	GRID	15	0	294.	28.	88.243350				
869-	GRID	16	0	287.	28.	91.2095				
870-	GRID	17	0	287.	28.	97.4755 0				
871-	GRID	18	0	294.	28.	100.44170				
872-	GRID	19	0	301.	28.	101.77850				
873-	GRID	20	0	308.	28.	101.27720				
874-	GRID	21	0	313.632840.		101.44550				
875-	GRID	22	0	312.933640.4663		88.103950				
876-	GRID	23		13.61865-7.		0.0				
877-	GRID	24		14.11995-14.		0.0				
878-	GRID	25		12.78315-21.		0.0				
879-	GRID	26		9.8165	-28.	0.0				
880-	GRID	27		3.5510	-28.	0.0				

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
881-	GRID	28		.58485	-21.	0.0				
882-	GRID	29		-0.75195	-14.	0.0				
883-	GRID	30		-0.25065	-7.	0.0				
884-	GRID	31		0.0	0.0	-21.1526				
885-	GRID	32		13.368	0.0	-21.1526				
886-	GRID	33		13.61865	-7.	-21.1526				
887-	GRID	34		14.11995	-14.	-21.1526				
888-	GRID	35		12.78315	-21.	-21.1526				
889-	GRID	36		9.8165	-28.	-21.1526				
890-	GRID	37		3.5510	-28.	-21.1526				
891-	GRID	38		.58485	-21.	-21.1526				
892-	GRID	39		-0.75195	-14.	-21.1526				
893-	GRID	40		-0.25065	-7.	-21.1526				
894-	GRID	41		0.0	0.0	-42.3051				
895-	GRID	42		13.368	0.0	-42.3051				
896-	GRID	43		13.51865	-7.	-42.3051				
897-	GRID	44		14.11995	-14.	-42.3051				
898-	GRID	45		12.78315	-21.	-42.3051				
899-	GRID	46		9.8165	-28.	-42.3051				
900-	GRID	47		3.5510	-28.	-42.3051				
901-	GRID	48		.58485	-21.	-42.3051				
902-	GRID	49		-0.75195	-14.	-42.3051				
903-	GRID	50		-0.25065	-7.	-42.3051				
904-	GRID	51		0.0	0.0	-68.8601				
905-	GRID	52		13.368	0.0	-68.8601				
906-	GRID	53		13.61865	-7.	-68.8601				
907-	GRID	54		14.11995	-14.	-68.8601				
908-	GRID	55		12.78315	-21.	-68.8601				
909-	GRID	56		9.8165	-28.	-68.8601				
910-	GRID	57		3.551	-28.	-68.8601				
911-	GRID	58		.58485	-21.	-68.8601				
912-	GRID	59		-0.75195	-14.	-68.8601				
913-	GRID	60		-0.25065	-7.	-68.8601				
914-	GRID	61		0.0	0.0	-95.4151				
915-	GRID	62		13.368	0.0	-95.4151				
916-	GRID	63		13.51865	-7.	-95.4151				
917-	GRID	64		14.11995	-14.	-95.4151				
918-	GRID	65		12.78315	-21.	-95.4151				
919-	GRID	66		9.8155	-28.	-95.4151				
920-	GRID	67		3.551	-28.	-95.4151				

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
921-	GRID	68		.58485	-21.	-95.4151				
922-	GRID	69		-0.75195	-14.	-95.4151				
923-	GRID	70		-0.25065	-7.	-95.4151				
924-	GRID	72		13.284	18.3566	-143.315		0		
925-	GRID	73		13.368	0.0	-121.775		0		
926-	GRID	74		13.61865	-7.	-121.775				
927-	GRID	75		14.11995	-14.	-121.775				
928-	GRID	76		12.78315	-21.	-121.775		0		
929-	GRID	77		9.8165	-28.	-121.775		0		
930-	GRID	78		3.551	-28.	-121.775		0		
931-	GRID	79		.58485	-21.	-121.775		0		
932-	GRID	80		-0.75195	-14.	-121.775				
933-	GRID	81		-0.25065	-7.	-121.775				
934-	GRID	82		0.0	0.0	-121.775		0		
935-	GRID	83		6.684	4.1966	-121.775		456		
936-	GRID	84		8.484	-23.3	-121.775		456		
937-	GRID	85		6.684	4.1966	-143.315		0		
938-	GRID	86		8.484	-23.3	-136.585		0		
939-	GRID	87		8.484	-23.3	-141.89		0		
940-	GRID	88	2	20.2	15.	16.8		0		
941-	GRID	89		6.684	4.1966	-146.435		0		
942-	GRID	90		2.084	12.1966	-143.315		0		
943-	GRID	91		14.334	8.3566	-143.315		0		
944-	GRID	92		13.368	0.0	-143.315		0		
945-	GRID	93		13.61865	-7.	-143.315		0		
946-	GRID	94		14.11995	-14.	-136.585		0		
947-	GRID	95		12.78315	-21.	-136.585		0		
948-	GRID	96		9.8165	-28.	-136.585		0		
949-	GRID	97		3.551	-28.	-136.585		0		
950-	GRID	98		.58485	-21.	-136.585		0		
951-	GRID	99		-0.75195	-14.	-136.585		0		
952-	GRID	100		-0.25065	-7.	-143.315		0		
953-	GRID	101		0.0	0.0	-143.315		0		
954-	GRID	102		3.494	18.3566	-143.315		0		
955-	GRID	103		5.084	25.6966	-143.315		0		
956-	GRID	104		12.334	25.6966	-143.315		0		
957-	GRID	105	0	100.	0.0	101.02658		123456		
958-	GRID	106		12.334	25.6996	-153.835				
959-	GRID	107		13.284	18.3566	-153.835				
960-	GRID	108		14.334	8.3556	-153.835		0		

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
961-	GRID	109		2.084	12.1966	-153.835	0			
962-	GRID	110		3.484	18.3565	-153.835				
963-	GRID	111		5.084	25.6966	-153.835				
964-	GRID	112		12.334	25.6995	-160.915				
965-	GRID	113		13.284	18.3566	-160.915				
966-	GRID	114		14.334	8.3556	-160.915	0			
967-	GRID	115		2.084	12.1966	-160.915	0			
968-	GRID	116		3.484	18.3566	-160.915				
969-	GRID	117		5.084	25.6966	-160.915				
970-	GRID	118		9.404	43.9466	-164.515	0			
971-	GRID	119		10.694	34.8765	-168.515	0			
972-	GRID	120		12.334	25.6995	-153.515	0			
973-	GRID	121		13.284	18.3566	-168.515	0			
974-	GRID	122		14.334	8.3556	-165.515	0			
975-	GRI:	123		2.094	12.1966	-164.515	0			
976-	GRID	124		3.484	18.3566	-168.515	0			
977-	GRID	125		5.084	25.6963	-153.515	0			
978-	.RID	126		5.694	34.8965	-168.515	0			
979-	GRID	127		6.484	43.9465	-154.515	0			
980-	GRID	128		12.309	43.9465	-173.572	0			
981-	GRID	129		9.404	43.9466	-177.82	0			
982-	GRID	130		10.584	34.8966	-177.82	0			
983-	GRID	131		12.334	25.6966	-177.82	0			
984-	GRID	132		13.284	18.3566	-177.82	0			
985-	GRID	133		14.334	8.3556	-177.82	0			
986-	GRID	134		6.694	4.1966	-177.82	0			
987-	GRID	135		2.094	12.1965	-177.92	0			
988-	GRID	136		3.494	18.3565	-177.82	0			
989-	GRID	137		5.084	25.6916	-177.82	0			
990-	GRID	1.8		-2.316	25.6966	-180.4	0			
991-	GRID	139		4.165	25.6968	-180.313	0			
992-	GRID	140		11.194	25.6965	-181.521	0			
993-	GRID	141		5.684	34.8966	-177.82	0			
994-	GRID	142		5.684	43.9466	-177.92	0			
995-	GRID	143		6.694	4.1966	-177.195	0			
996-	GRIC	144	2	10.	146.667	5.68	0			
997-	GRID	145	2	10.	118.333	5.68	0			
998-	GRID	146	2	10.	90.	5.68	0			
999-	GRID	147	2	10.	61.667	5.68	0			
1000-	GRID	148	2	10.	33.333	5.68	0			

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
1001-	GRID	149	2	14.	8.13	5.68		0		
1002-	GRID	150	2	14.	351.87	5.68		0		
1003-	GRID	151	2	10.	5.	5.68		0		
1004-	GRID	152	2	10.	333.333	5.68		0		
1005-	GRID	153	2	13.	301.667	5.68		0		
1006-	GRID	154	2	10.	270.	5.68		0		
1007-	GRID	155	2	10.	238.333	5.68		0		
1008-	GRID	156	2	10.	206.667	5.68		0		
1009-	GRID	157	2	14.	188.13	5.68		0		
1010-	GRID	158	2	14.	171.87	5.68		0		
1011-	GRID	159	2	7.3	131.5	5.68		0		
1012-	GRID	160	2	5.45	90.	5.68		0		
1013-	GRID	161	2	7.3	48.5	5.68		0		
1014-	GRID	162	2	5.05	10.2	5.68		0		
1015-	GRID	163	2	6.8	318.75	5.68		0		
1016-	GRID	164	2	4.5	270.	5.68		0		
1017-	GRID	165	2	.9	90.	5.68		0		
1018-	GRID	166	2	5.05	169.8	5.68		0		
1019-	GRID	167	2	6.8	221.25	5.68		0		
1020-	GRID	168	2	10.	175.	5.68		0		
1021-	GRID	169	2	10.	175.	9.93		0		
1022-	GRID	170	2	10.	146.667	9.93		0		
1023-	GRID	171	2	10.	118.333	9.93		0		
1024-	GRID	172	2	10.	90.	9.93		0		
1025-	GRID	173	2	10.	61.667	9.93		0		
1026-	GRID	174	2	10.	33.333	9.93		0		
1027-	GRID	175	2	10.	5.	9.93		0		
1028-	GRID	176	2	10.	333.333	9.93		0		
1029-	GRID	177	2	10.	301.667	9.93		0		
1030-	GRID	178	2	10.	270.	9.93		0		
1031-	GRID	179	2	10.	238.333	9.93		0		
1032-	GRID	180	2	10.	206.667	9.93		0		
1033-	GRID	181	2	10.	206.667	14.18		0		
1034-	GRID	182	2	10.	175.	14.18		0		
1035-	GRID	183	2	10.	146.667	14.18		0		
1036-	GRID	184	2	10.	118.333	14.18		0		
1037-	GRID	185	2	10.	90.	14.18		0		
1038-	GRID	186	2	10.	61.667	14.18		0		
1039-	GRID	187	2	10.	33.333	14.18		0		
1040-	GRID	188	2	10.	5.	14.18		0		

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 .. 9 .. 10 .
1041-	GRID	189	2	10.	333.333	19.18	0	
1042-	GRID	190	2	10.	301.667	19.18	0	
1043-	GRID	191	2	10.	270.	19.18	0	
1044-	GRID	192	2	10.	238.333	19.18	0	
1045-	GRID	193	2	10.	238.333	18.93	0	
1046-	GRID	194	2	10.	206.667	18.93	0	
1047-	GRID	195	2	10.	175.	18.93	0	
1048-	GRID	196	2	10.	146.667	18.93	0	
1049-	GRID	197	2	10.	118.333	18.93	0	
1050-	GRID	198	2	10.	90.	18.93	0	
1051-	GRID	199	2	10.	61.667	18.93	0	
1052-	GRID	200	2	10.	33.333	18.93	0	
1053-	GRID	201	2	10.	5.	18.93	0	
1054-	GRID	202	2	10.	333.333	18.93	0	
1055-	GRID	203	2	10.	301.667	18.93	0	
1056-	GRID	204	2	10.	270.	18.93	0	
1057-	GRID	205	2	10.	270.	23.68	0	
1058-	GRID	206	2	10.	238.333	23.68	0	
1059-	GRID	207	2	10.	206.667	23.68	0	
1060-	GRID	208	2	10.	175.	23.68	0	
1061-	GRID	209	2	10.	146.667	23.68	0	
1062-	GRID	210	2	10.	118.333	23.68	0	
1063-	GRID	211	2	10.	90.	23.68	0	
1064-	GRID	212	2	10.	61.667	23.68	0	
1065-	GRID	213	2	10.	33.333	23.68	0	
1066-	GRID	214	2	10.	5.	23.68	0	
1067-	GRID	215	2	10.	333.333	23.68	0	
1068-	GRID	216	2	10.	301.667	23.68	0	
1069-	GRID	217	2	10.	301.667	35.18	0	
1070-	GRID	218	2	10.	270.	35.18	0	
1071-	GRID	219	2	10.	238.333	35.18	0	
1072-	GRID	220	2	10.	206.667	35.18	0	
1073-	GRID	221	2	10.	175.	35.18	0	
1074-	GRID	222	2	10.	146.667	35.18	0	
1075-	GRID	223	2	10.	118.333	35.18	0	
1076-	GRID	224	2	10.	90.	35.18	0	
1077-	GRID	225	2	10.	61.667	35.18	0	
1078-	GRID	226	2	10.	33.333	35.18	0	
1079-	GRID	227	2	10.	5.	35.19	0	
1080-	GRID	228	2	10.	333.333	35.18	0	

.ILT ROTOR NORMAL MODES ANALYSIS　　　　FEBRUARY 20, 1978　NASTRAN 4/ 1/76　PAGE 38
SYMMETRIC MODES CRUISE ATTITUDE

S O R T E D　B U L K　D A T A　E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 .. 10 .
1081-	GRID	229	2	0.0	0.0	35.1[illegible]		0	
1082-	GRID	230	0	[illegible]	199.	100.314	0	0	
1083-	GRID	231		6.[illegible]84	4.1966	-159.322		123456	
1084-	GRID	240	2	20.	0.0	0.0		123456	
1085-	GRID	241		0.0	0.0	0.0		123456	
1086-	GRID	242		5.684	34.8966	-160.905			
1087-	GRID	243		10.684	34.8965	-160.905			
1088-	GRID	244		6.201	38.442	-160.905		0	
1089-	GRID	245		10.288	38.442	-160.905		0	
1090-	GRID	246		5.752	33.315	-153.835		0	
1091-	GRID	247		11.111	33.315	-153.835		0	
1092-	GRID	248	0	329.42	231.25	101.809		0	
1093-	GRID	249	0	336.17	231.62	101.809		0	
1094-	GRID	250	0	317.02	229.50	110.809		0	
1095-	GRID	251	0	336.17	231.62	96.270		0	
1096-	GRID	252	0	291.33	0.0	50.064	0	246	
1097-	GRID	253	0	555.4	0.0	133.0	0	0	
1098-	MAT1	1	1.E7	.4E7		.101			
1099-	MAT1	2	2.9E7	1.1E7		.283			
1100-	MAT1	3	1.E7	.4E7		.0			
1101-	PARAM	GRDPNT	252						
1102-	PARAM	WTMASS	.002587						
1103-	PBAR	1	1	1.623	.8393	.8393	1.6786		
1104-	PBAR	2	1	3.499	4.1136	4.1136	8.2272		
1105-	PBAR	3	1	1.0	.333	.0208	.0702		
1106-	PBAR	4	1	2.0	1.13	1.13	.10		
1107-	PBAR	5	1	1.4688	.563[illegible]	.5634	.00748		
1108-	PBAR	6	1	2.10	.94	.94	.10		
1109-	PBAR	7	1	5.905	4.2895	4.9129	3.57		
1110-	PBAR	8	1	12.054	8.274	12.719	19.107		
1111-	PBAR	9	2	2.843	6.032	6.032	12.064		
1112-	PBAR	10	2	5.698	24.127	24.127	48.254		
1113-	PBAR	11	1	5.5	10.985	13.375	.4583		
1114-	PBAR	12	1	1.01	1.303	.1585	.0119		
1115-	PBAR	13	1	.625	.8138	.013	.039		
1116-	PBAR	14	1	2.84	2.36	2.36	.125		
1117-	PBAR	15	1	.56	.0741	.0741	.00195		
1118-	PBAR	16	1	4.273	154.37	154.37	308.74		
1119-	PBAR	17	1	8.98	6.28	8.82	11.34		
1120-	PBAR	18	3	10.0	5000.	5000.	10000.		

S O R T E D B U L K D A T A E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
1121-	PBAR	19	3	4.0	150.	150.	150.			
1122-	PBAR	20	1	.25	.05	.05	.0009			
1123-	PLOTEL	1500	89	143		1501	229	231		
1124-	PQUAD2	8	1	.05						
1125-	PSHEAR	1	1	.25						
1126-	PSHEAR	2	1	.16						
1127-	PSHEAR	3	1	.01						
1128-	PSHEAR	4	1	.025						
1129-	PSHEAR	5	1	.04						
1130-	PSHEAR	6	1	.071						
1131-	PSHEAR	7	1	.063						
1132-	PSHEAR	8	1	.10						
1133-	PTRIA2	1	1	.050						
1134-	SEQGP	1	3	252	2	2	4	6	12	
1135-	SEQGP	7	11	253	1	84	149	86	162	
1136-	SEQGP	9	7	5	10	8	9	13	17	
1137-	SEQGP	14	19	15	21	19	20	19	18	
1138-	SEQGP	17	22	21	26	22	27	26	35	
1139-	SEQGP	20	16	23	29	24	31	25	33	
1140-	SEQGP	27	34	31	44	32	45	36	53	
1141-	SEQGP	28	32	29	30	30	28	33	47	
1142-	SEQGP	34	43	35	51	39	50	39	48	
1143-	SEQGP	37	52	41	66	42	65	46	74	
1144-	SEQGP	40	46	43	68	44	70	45	72	
1145-	SEQGP	47	73	51	89	52	90	56	100	
1146-	SEQGP	48	71	49	69	50	67	53	93	
1147-	SEQGP	54	96	53	98	58	97	59	95	
1148-	SEQGP	57	99	61	91	62	92	66	126	
1149-	SEQGP	60	94	63	116	64	121	65	124	
1150-	SEQGP	67	125	3	6	10	5	4	8	
1151-	SEQGP	68	123	69	122	70	115	82	113	
1152-	SEQGP	73	114	74	118	75	135	76	138	
1153-	SEQGP	77	141	78	140	79	139	80	134	
1154-	SEQGP	81	117	109	111	115	85	108	109	
1155-	SEQGP	83	133	134	142	137	78	131	77	
1156-	SEQGP	87	181	97	148	95	146	96	147	
1157-	SEQGP	91	129	102	112	72	110	85	143	
1158-	SEQGP	93	120	101	131	92	132	90	[illegible]	
1159-	SEQGP	98	145	99	136	94	137	100	[illegible]	
1160-	SEQGP	104	87	248	36	250	37	140	54	

S O R T E D   B U L K   D A T A   E C H O

CARD COUNT	. 1 ..	2 ..	3 ..	4 ..	5 ..	6 ..	7 ..	8 ..	9 ..	10 .
1161-	SEQGP	113	81	106	82	112	79	111	86	
1162-	SEQGP	114	84	110	108	116	83	107	107	
1163-	SEQGP	117	80	163	159	164	158	165	144	
1164-	SEQGP	120	75	125	76	126	60	119	59	
1165-	SEQGP	123	136	135	128	122	103	133	127	30 18
1166-	SEQGP	124	102	121	101	132	104	136	105	
1167-	SEQGP	138	58	249	24	251	13	128	25	
1168-	SEQGP	141	56	127	41	118	40	130	55	
1169-	SEQGP	142	39	129	38	139	57	229	240	
1170-	SEQGP	143	153	144	163	156	170	157	166	
1171-	SEQGP	147	167	153	176	154	175	155	173	
1172-	SEQGP	152	174	148	168	89	196	188	199	
1173-	SEQGP	158	164	89	156	149	169	150	.72	
1174-	SEQGP	160	151	161	154	242	62	243	61	
1175-	SEQGP	162	155	166	152	167	157	159	150	
1176-	SEQGP	169	182	170	179	176	187	175	185	
1177-	SEQGP	172	177	173	180	177	189	178	188	
1178-	SEQGP	174	184	151	171	168	165	244	43	
1179-	SEQGP	179	186	171	178	145	160	146	161	
1180-	SEQGP	182	194	183	193	184	192	185	191	
1181-	SEQGP	186	190	201	210	199	204	180	183	
1182-	SEQGP	190	202	191	201	192	198	181	197	
1183-	SEQGP	194	209	195	208	196	207	197	206	
1184-	SEQGP	198	205	200	203	187	195	189	200	
1185-	SEQGP	202	213	203	214	204	212	193	211	
1186-	SEQGP	208	223	209	222	210	220	211	219	
1187-	SEQGP	212	217	213	216	214	215	215	228	
1188-	SEQGP	216	230	205	229	206	226	207	225	
1189-	SEQGP	217	238	219	237	221	236	223	235	
1190-	SEQGP	220	227	222	224	224	221	226	218	
1191-	SEQGP	225	234	227	233	228	232	218	231	
1192-	SEQGP	230	239	11	14	12	15	16	23	
1193-	SEQGP	245	42	246	64	247	63	103	88	
	ENDDATA									

NO ERRORS FOUND - EXECUTE NASTRAN PROGRAM

*** USER WARNING MESSAGE 2015, NO ELEMENTS CONNECT INTERNAL GRID POINT 105

*** USER WARNING MESSAGE 2015, NO ELEMENTS CONNECT INTERNAL GRID POINT 233

*** USER WARNING MESSAGE 2015, NO ELEMENTS CONNECT INTERNAL GRID POINT 243

*** USER WARNING MESSAGE 2015, NO ELEMENTS CONNECT INTERNAL GRID POINT 244

*** SYSTEM INFORMATION MESSAGE 3113, EMGPRO PROCESSING DOUBLE PRECISION ELEMENTS OF TYPE 34 STARTING WITH ID 278